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USAF DAMAGE TOLERANT DESIGN HANDBOOK: GUIDELINES FOR THE ANALYSIS AND DESIGN OF DAMAGE TOLERANT AIRCRAFT

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This is the first edition of a handbook to support the USAF Airplane Damage Tolerance Requirements contained in MIL-A-83444. The handbook provides specific background data and justification for the detailed requirements of MIL-A-83444 and provides guidelines and state-of-the-art analysis methods to assist contractor and USAF personnel in complying with the intent of the specification and in solving cracking problems, in general, for metallic aircraft structures. The material ~~contained~~ in this document is general

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enough to be useful in the evaluation of the damage tolerance of in-service aircraft designed and qualified prior to the issuance of MIL-A-83444. The handbook was ~~has been~~ structured to provide a clear and concise summary of the specification, MIL-A-83444, as well as supporting analysis methods, test techniques, and nondestructive inspection (NDI) methods are provided as state-of-the-art along with suggested and/or recommended practices, limitations, etc. ~~For the convenience of the user,~~ copies of appropriate USAF structural specifications are contained as an appendix to this handbook.

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DAMAGE TOLERANT DESIGN HANDBOOK

Guidelines for the Analysis and Design of Damage Tolerant Aircraft

FOREWORD

This report was prepared by Howard A. Wood and Robert M. Engle Jr. of the Structural Integrity Branch, Structures and Dynamics Division, Air Force Flight Dynamics Laboratory. The work was performed in-house under Work Unit 24010109, "Life Analysis and Design Methods for Aerospace Structures."

Major portions of Chapter 4, "Determination of Residual Strength" and Chapter 5, "Analysis of Damage Accumulation" were developed by Battelle Memorial Institute, Columbus, Ohio under Contract F33615-75-C-3101. The principal authors for Battelle were D. Broek and S. H. Smith. The contract was administered by R. M Engle Jr., Project Engineer, AFFDL/FBE.

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1. USAF solicits your comments concerning this handbook so that its usefulness may be improved in later editions. Send any comments to the following address:

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2. Comments are solicited in the following areas:

   a. Is the handbook adequate?
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3. Please note any specific errors which have been discovered. Include the page number for reference.

4. Revision B of the handbook is already underway so early submission of any comments would be greatly appeciated.

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CHAPTER 1

INTRODUCTION

1.0 INTRODUCTION

1.1 GENERAL

This is the first edition of a document to support the USAF Airplane Damage Tolerance Requirements contained in MIL-A-83444. The purpose of the handbook is to provide specific background data and justification for the detailed requirements of MIL-A-83444 and to provide guidelines and state-of-the-art analysis methods to assist contractor and USAF personnel in complying with the intent of the specification as well as in the solution of cracking problems, in general, for metallic aircraft structures.

The material contained in this volume is general enough to be useful in the evaluation of the damage tolerance of in-service aircraft designed and qualified prior to the issuance of MIL-A-83444. Damage tolerance analyses require supporting fracture mechanics materials data. A primary source of such data is MCIC-HB-01, "A Compilation of Fracture and Crack Growth Data for High Strength Alloys" which contains current data such as critical plane-strain stress-intensity factors (K_{IC}), plane stress and transitional stress-intensity factors (K_C), threshold stress-intensity factors in corrosive media (K_{ISCC}), sustained load crack growth rates in corrosive media (da/dt vs K_I) and fatigue crack growth rates (da/dN vs ΔK). For the convenience of the user, copies of appropriate USAF structural specifications are contained as an appendix to this handbook. Any conflict or discrepancy in information contained in this handbook and/or MIL-A-83444 is unintentional and in all cases, the governing document is the current version of the specifications. Throughout the handbook references to MIL-A-8344 will be specified as paragraph numbers in parentheses, e.g. (3.1.1.2) refers to paragraph 3.1.1.2 <u>Continuing damage</u> on page 4 of MIL-A-83444.

1.2 BACKGROUND

Structural failures and cracking problems encountered on various military aircraft have contributed to overcost, behind schedule airframe developments, unacceptably high in-service maintenance and repair costs, excessive downtimes, and, in some cases, loss of life. Air Force reviews of these problems have led to the conclusion that fatigue, stress corrosion, and corrosion-fatigue are the primary mechanisms of crack growth. In addition, it has been found that pre-existing manufacturing quality deficiencies (e.g. scratches, flaws, burrs, cracks, etc.) or service induced damage (e.g. corrosion pits), are very often the basic cause of the cracking problems. The effect of these flaws on the safety of the aircraft is dependent on their initial sizes, the rates of growth with service usage, the critical flaw sizes, the inspectability of the structure, and the fracture containment capabilities of the basic structural design. <u>From the standpoint of flight safety, it is prudent to assume that new airframe structures can and very often do contain such initial damage</u>. Likewise, for older systems and those structures which have experienced service cracking, it is essential that safety of flight be provided through the consideration of an "initial flaw" model in which some size of initial damage is assumed to exist consistent with the inspection capability either in the field or during manufacture. The critical assumed damage shall be that considered to be just smaller than can be detected by the appropriate NDI methods.

It is the intent of MIL-A-83444 to ensure that the maximum possible initial damage will not grow to a size such as to endanger flight safety anytime during the design life of the aircraft. When properly interpreted and applied, the specification requirements should accomplish this intent through:

a. Proper material selection and control

b. Control of stress levels

c. Use of fracture resistant design concepts

d. Manufacturing process control

e. Use of qualified inspection procedures

While it is expected that compliance with the requirements will also tend to lead to improved structural durability, this is not their primary purpose. Requirements directed towards minimizing and delaying crack initiation and structural deterioration due to fatigue and corrosion, i.e. durability, are contained in MIL-A-8866B.

1.3 HANDBOOK DESCRIPTION

The current document has been structured to provide a clear and concise summary of the specification, MIL-A-83444, as well as supporting data and rationale behind the critical assumptions. Where appropriate, analysis methods, test techniques, and NDI methods are provided as state-of-the-art with suggested and/or recommended practices, limitations, etc., so stated. Chapters 2 through 11 address the following topics:

Chapter 2.0 - Summary of Requirements contains a review of MIL-A-83444 including examples for clarity, data to support specific requirements, and assumptions and rationale where limited data exists.

Chapter 3.0 - Damage Size Considerations discusses appropriate NDI practice, state-of-the-art procedures, demonstration programs to qualify NDI, in service NDI practice and specific examples illustrating how damage is assumed to exist in structures.

Chapter 4.0 - Determination of Residual Strength summarizes theory, methods, assumptions required, material data, test verification, and examples to estimating the final fracture strength or crack arrest potential of cracked structures.

Chapter 5.0 - Analysis of Damage Growth describes current practice of estimating the rate of crack growth as a function of time, cyclic and sustained load occurrence; gives examples indicating limitations of methods, use of material data and suggested testing to support predictions and establish confidence.

Chapter 6.0 - Damage Tolerance Analysis - Sample Problems - provides detailed analysis of typical structural examples illustrating methodology and assumptions required.

Chapter 7.0 - Damage Tolerance Testing describes methods and recommended tests to verify methods, full-scale testing to verify residual strength and slow crack growth rates.

Chapter 8.0 - Individual Airplane Tracking describes current methods available to account for usage variations for individual force aircraft based on a crack growth model.

Chapter 9.0 - Fracture Control Guidelines describes methods and procedures for development and implementation of a damage tolerance control plan as required in MIL-STD 1530 (5.1.3.1).

Chapter 10.0 - Design Guidelines describes the factors which should be considered when designing new structure to meet the requirements of MIL-A-83444.

Chapter 11.0 - Repair Guidelines describes the factors which should be considered when designing a repair, in order to ensure that the basic damage tolerance present in the original structure is not degraded by the repair.

CHAPTER 2

SUMMARY OF REQUIREMENTS

2.0 SUMMARY OF REQUIREMENTS

SUMMARY OF REQUIREMENTS
(Cont'd)

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Definitions

Minimum Assumed Initial Damage Size (Section 2.4) - The minimum assumed initial damage size is the smallest crack-like defect which shall be used as a starting point for analyzing residual strength and crack growth characteristics of the structure.

Minimum Assumed In-Service Damage Size (Section 2.5) - The minimum assumed in-service damage size is the smallest damage which shall be assumed to exist in the structure after completion of an in-service inspection.

Minimum Period of Unrepaired Service Usage (Section 2.7) - The minimum period of unrepaired service usage is that period of service time during which the appropriate level of damage (assumed initial or in-service) is presumed to remain unrepaired and allowed to grow within the structure.

Minimum Required Residual Strength Load (Section 2.6) - The minimum required residual strength is specified as the smallest internal member load which the aircraft must be able to sustain with damage present and without endangering safety of flight or degrading the performance of the aircraft for the specified minimum period of unrepaired service usage.

Damage Size Growth Limit - The damage size growth limit is the maximum size to which initial or in-service size damage is allowed to grow without degrading the residual strength level below its required level.

2.1 GENERAL

USAF damage tolerance design requirements as specified in MIL-A-83444 apply to all safety of flight structure, that is, structure whose failure could cause direct loss of the aircraft, or whose failure, if it remained undetected, could result in the loss of aircraft. The requirements stipulate that damage is assumed to exist in each element of new structure in a conservative fashion (i.e., critical orientation with respect to stress field and in a region of highest stress). The structure must successfully contain the growth of the initial assumed damage for a specified period of service while maintaining a minimum level of residual static strength, both during and at the end of this period. Figure 2.1* illustrates these requirements in diagrammatic form. Since residual static strength generally decreases with increased damage size, the residual strength and growth requirements are coupled with the maximum allowable damage size or damage size growth limit established by the minimum-required residual strength load. The safe growth period (period of unrepaired service usage) is coupled to either the design life requirement for the air vehicle or to the scheduled in-service inspection intervals. While the specific requirements of MIL-A-83444 may seem more complex than described in Figure 2.1, all essential elements are as illustrated. The remainder of Chapter 2 will describe these individual elements.

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*Figures for Chapter 2 are located at the end of Chapter 2.

A structure can be qualified under one of two categories of defined damage tolerance (referred to as Design Concepts in MIL-A-83444). These are:

> Slow Crack Growth - In this category, structures are designed such that initial damage will grow at a stable, slow rate under service environment and not achieve a size large enough to cause rapid unstable propagation.
>
> Fail Safe - In this category, structures are designed such that propagating damage is safely contained by failing a major load path or by other damage arrestment features.

In Slow Crack Growth qualified structure, damage tolerance (and thus safety) is assured only by the maintenance of a slow rate of growth of damage, a residual strength capacity and the assurance that subcritical damage will either be detected at the depot or will not reach unstable dimensions within several design life times. In Fail Safe qualified structure, damage tolerance (and thus safety) is assured by the allowance of partial structural failure, the ability to detect this failure prior to total loss of the structure, the ability to operate safely with the partial failure prior to inspection and the maintenance of specified static residual strength throughout this period.

MIL-A-83444 requirements have been developed with the intention of providing approximately the same level of damage tolerance for the

Slow Crack Growth and Fail Safe categories. As discussed in Section 2.4, this is accomplished mainly by varying the assumed initial flaw sizes.

Each structure must qualify within one of the designated categories of in-service inspectability (referred to as "The Degree of Inspectability" in MIL-A-83444), including the option to designate Slow Crack Growth qualified structure as "in-service non-inspectable." The various degrees of inspectability refer to methods, equipment, and other techniques for conducting in-service inspections as well as accessibility and the location of the inspection (i.e., field or depot) and are defined in Section 6.2 of MIL-A-83444.

In the specification, the detailed requirements are grouped according to the particular design category:

Slow Crack Growth: Section 3.2.1

Fail Safe: multiple load path - Section 3.2.2

crack arrest - Section 3.2.3

The selection of the most appropriate damage tolerance category under which to qualify the structure is the choice of the designer/analyst. The choice of degree of in-service inspectability is somewhat limited, however, to those described in MIL-A-83444. The inspection requirements have been developed based upon past and present experiences and are felt to be reasonable estimates of future practice.

It is the intent of this specification to provide for at least design limit load residual strength capability for all intact structure (i.e., for subcritical damage sizes in slow crack growth structure and damage sizes less than a failed load path in fail safe qualified designs). This requirement allows for full limit load design capability and thus unrestricted aircraft usage. The imposition of the requirement constrains structure qualified as Slow Crack Growth to either depot level inspectable or depot level non-inspectable.

To help in understanding the steps required to utilize MIL-A-83444, the essential elements and the corresponding paragraphs required for the Slow Crack Growth and Fail Safe categories are indicated in Figure 2.2. Each vertical path describes the sequence to be followed to check a structure for the appropriate category. As described in Section 2.2, fail safe structure must meet both the intact structure and remaining structure requirements. Slow crack growth structure will meet either the depot level inspectable or the non-inspectable structure requirements. For each structure evaluation of the following parameters are required:

a. Design Category - Optional Choice of Designer

b. Degree of In-Service Inspectability - Types of Inspection defined - selection of category is program option.

c. Inspection Intervals - Values specified for various categories - should be used in design but may be altered for specific design based on individual system needs.

d. Initial Damage, In-Service Damage and Continuing Damage Assumptions - Values specified - alternate values allowed if justified and demonstrated (See Section 2.4.4).

e. Minimum Required Residual Strength - Means of obtaining value specified in terms of inspection categories and inspection intervals - no options provided.

f. Damage Size Growth Limits - Defined

g. Periods of Unrepaired Service Usage - Specified

h. Remaining Structure Damage Sizes - Defined

2.2 DESIGN CATEGORY

Selection of appropriate design category (e.g. Fail Safe or Slow Crack Growth) (Figure 2.3) is the initial step in applying MIL-A-83444. In the development of the specification it was recognized that multiple load path and crack arrest type structure have inherent potential for tolerating damage by virtue of geometric design features. On the other hand, it is often not possible to avoid primary structure with only one major load path and some provisions are necessary to insure that these situations can be designed to be damage tolerant. It is the intent of the specification to encourage the exploration of the potentials for damage tolerance in each type of structure. Single load path or monolithic structures must rely on the slow rate of growth of damage for safety and thus, the design stress level and material selection become the controlling factors.

While single load path "monolithic" structures must be qualified as Slow Crack Growth, the designer has the choice of category for qualification of multiple load path cases. The decision may be made to qualify multiple load path structure as Slow Crack Growth for various reasons such as the inability to meet portions of the requirement for Fail Safe (e.g. Remaining Structure Damage Growth Residual Strength) or because the job of conducting a slow crack growth analysis is less complex. The specification allows this flexibility. The important factors to consider when deciding on such options are: (1) the method of construction utilized is not synonomous with design category selected (if all multiple load path structure is not fail safe) and (2) once a category is chosen, the structure must meet all the requirements in the specification that cover that category.

The mere fact that a structure has alternate load paths (local redundancy) in some locations does not allow it to be qualified as Fail Safe. Some examples are helpful in illustrating this point:

EXAMPLE 1

The fitting illustrated in Figure 2.4 has multiple lug ends at the pinned connection. Failure or partial failure of one of the lugs (A) would allow the load to be redistributed to the sound structure. Localized redundancy is often beneficial and in this case is good design practice. However, the fitting cannot be qualified as multiple load path structure since the occurrence and growth of damage at a typical location (B) would render the structure inoperative. The only means of protecting the safety of this structure would be to qualify it as Slow Crack Growth.

EXAMPLE 2

In this example (Figure 2.5), a wing box is attached to the fuselage carry through structure by multiple fittings. Upper and lower skin are one piece for manufacturing and cost reduction. Substructure consists of multiple spars spaced to attach to the individual attachment fittings.

A case could be made to qualify this structure as fail safe multiple load path. Depending upon the amount of bending carried by the spars, it would be possible to design such that damage in the skin would be arrested at a spar prior to becoming critical. The design might also tolerate failure of one spar cap and a portion of the skin, prior to catastrophic failure.

The attachment system could be designed to satisfy fail safe requirements with one fitting failed. On the other hand, if the skin was the major bending member and the design stress of sufficient magnitudes to result in a relatively short critical crack length, then the skin and spar structure could only be qualified as slow crack growth structure.

These examples illustrate the fact that structure is often locally redundant (usually good design practice), but in an overall sense may not be able to take advantage of this redundancy to be qualified Fail Safe. Considerable judgement is required for the selection of potential initial damage locations for the assessment of damage growth patterns and the selection of major load paths. The qualification as fail safe is thus a complex procedure entailing judgment and analysis. Because of this, the choice is often made to qualify the design as slow crack growth regardless of the type of construction.

2.3 INSPECTION CATEGORIES AND INSPECTION INTERVALS

Descriptions of degree of inspection are contained in Section 6.2.1 of MIL-A-83444. This information is reproduced in Table 2-1.

For each individual aircraft system, the Air Force is obligated to specify the planned major depot and base level inspection intervals to be used in the design of the aircraft. Typically these intervals will be approximately 1/4 of the design service life. The types and extent of inspection (i.e., equipment, accessibility, necessity for part removal, etc.) required at each of these major inspections is dependent upon the specific aircraft design and how the results of development and full-scale tests and/or service experience may have modified the original plans. The Air Force desire is for the contractor to design damage tolerant structure, which will minimize the need for extensive non-destructive depot and/or base level inspections. Thus, primary emphasis should be placed on obtaining designs for which significant damage sizes can be found readily by visual inspection. However, where periodic inspections are required in order to satisfy the damage tolerance requirements, the contractor must recognize that the USAF will most likely be conducting the inspection and thus, the inspection categories of MIL-A-83444 reflect this capability.

The design of some specific aircraft components for intermediate special visual inspections (6.2.1.4) (typically once per year) may be advantageous from a performance and/or cost standpoint and may be used by the

TABLE 2-1 Summary of In-Service Inspection

Degree of Inspectability	Typical Inspection Interval
In-Flight evident inspectable. Structure is in-flight evident inspectable if the nature and extent of damage occurring in flight will result directly in characteristics which make the flight crew immediately and unmistakably aware that significant damage has occurred and that the mission should not be continued.	One Flight
Ground evident inspectable. Structure is ground evident inspectable if the nature and extent of damage will be readily and unmistakably obvious to ground personnel without specifically inspecting the structure for damage.	One Flight
Walkaround inspectable. Structure is walkaround inspectable if the nature and extent of damage is unlikely to be overlooked by personnel conducting a visual inspection of the structure. This inspection normally shall be a visual look at the exterior of the structure from ground level without removal of access panels or doors without special inspection aids.	Ten Flights
Special visual inspectable. Structure is special visual inspectable if the nature and extent of damage is unlikely to be overlooked by personnel conducting a detailed visual inspection of the aircraft for the purpose of finding damaged structure. The procedure may include removal of access panels and doors, and may permit simple visual aids such as mirrors and magnifying glasses. Removal of paint, sealant, etc., and use of NDI techniques such as penetrant, X-ray, etc. are not part of a special visual inspection.	One Year

TABLE 2-1 Summary of In-Service Inspection (continued)

Depot or base level inspection. Structure is depot or base level inspectable if the nature and extent of damage will be detected utilizing one or more selected nondestructive inspection procedures. The inspection procedures may include NDI techniques such as penetrant, X-ray, ultrasonic, etc. Accessibility considerations may include removal of those components designed for removal.	1/4 Design Service Lifetime
In-Service non-inspectable structure. Structure is in-service non-inspectable if either damage size or accessibility preclude detection during one or more of the above inspections.	One design service Lifetime

contractor in satisfying the requirements. Normally, special visual inspections will not be specified by the Air Force in the design/ development stage but may be dictated, subsequent to design, by the results of testing and/or service experience.

Other visual inspectability levels include the categories of walk-around inspectable (6.2.1.3), ground evident (6.2.1.2), and in-flight evident (6.2.1.1). These inspections generally do not involve either significant cost or time but can play a major role in the maintenance of aircraft safety.

The assumed Air Force depot or base level inspection capabilities depend on the type of inspection performed. In those special cases where the potential benefits justify it, the contractor may assume during design and recommend to the Air Force that specific components be removed from the aircraft and inspected during scheduled depot or base level inspections. In these cases, the assumed initial damage sizes subsequent to the inspection shall be the same as those in the original design providing the same inspection procedures are used and certified inspection personnel perform the inspection.

Conventional NDI procedures such as X-ray, penetrant, magnetic particle, ultrasonic, and eddy current are generally available for depot or base level inspections and will be performed as dictated by the specific aircraft design inspection requirements and as they may have been modified by subsequent tests and service experience. In establishing the design inspection requirements, the contractor should attempt to minimize the need for such NDI and should not plan on (or design for) general fastener pulling inspections. The specified frequency of inspections for each of the inspectability levels is indicated in 3.2.2.1 of MIL-A-83444 and Table 2-1 and represents estimates of typical inspection intervals only. As previously mentioned, the typical depot or base level frequency is once every one quarter of the design lifetime but may be otherwise specified in the appropriate contractual document. Special visual requires Air Force approval before being considered as a design constraint but shall not be required more frequently than once per year. Again, the justification for this restriction is cost and schedule requirements.

2.4 INITIAL DAMAGE ASSUMPTIONS

2.4.1 Intact Structure Primary Damage Assumption - The basic premise in arriving at the initial damage sizes is the assumption that the as-fabricated structure contains flaws of a size just smaller than the non-destructive (NDI) maximum undetectable flaw size. However, for any non-destructive inspection procedure/material/structure combination, the maximum undetectable flaw size can only be specified in a meaningful manner if the probability of detecting that flaw and the confidence level associated with the probability are also specified. MIL-A-83444 requires that the probability of detection and confidence levels be 90% and 95%, respectively, for the slow crack growth category and 90% and 50%, respectively, for the fail safe category (see Figure 2.6). The 90%-95% values were selected as being economically practical from the standpoint of performing a non-destructive test demonstration program (see also Chapter 3.0). The 90%-95% is also the basis for MIL-HDBK-5 for "B" allowable values. The same probability of detection value (i.e., 90%) is specified for the fail safe category as for the slow crack growth category since NDI capability is not category dependent in the sense specified in MIL-A-83444. Because of the fracture containment capabilities and required in-service inspectability of the fail safe category, it appears reasonable to accept a lower confidence level on detectability. A somewhat arbitrary value of 50% is specified. This, in effect, results in a smaller value of required initial flaw size assumption for the intact structure requirement of the Fail Safe category than for the Slow Crack Growth category. As a result of planned studies, the specified reliability and confidence levels may be changed in future revisions to MIL-A-83444. Figure 2.7 summarizes the initial damage assumptions for intact structure as specified in MIL-A-83444.

Typical types of manufacturing damage which have been seen on past military aircraft programs are shown in Figure 2.8. These flaw size shapes which are intended to be covered by the initial flaw size assumptions include radial tears, cracked burrs and rifle marks at fastener holes as well as forging defects, welding defects, heat treatment cracks, forming cracks, and machinery damage at locations other than fastener holes.

Based on a review of existing NDI data, the values of 0.050 and 0.020 were selected as most appropriate to be specified for the slow crack growth and fail safe categories, respectively. The contractor is given the option of demonstrating better inspection capability to specified probability and confidence levels (see Chapter 3.0). The 0.050" crack size for holes and cutouts is based on NDI reliability data obtained using eddy current inspection with fastener removed. (See, for example, Figure 2.9). The surface flaw size 0.250" in length by .125" in depth was obtained from Air Force sponsored inspection reliability programs where several techniques were used including ultrasonic, dye penetrant and magnetic particle (Figure 2.10). In these programs, most techniques were found to be sensitive to both surface length and flaw depth and thus the NDI capability must be judged in terms of the flaw shape parameter a/Q* rather than simply surface length or crack depth. For 90% probability, 95% confidence an approximate value of a/Q = .05 was established as applicable for the slow crack growth category. However, for ease of analysis, it was decided to specify the dimensions of a semicircular crack whose a/Q = .050. This resulted in the ".125 x .25" surface flaw for the slow crack growth category.

For fail safe structure inspection to 90% probability, 50% confidence results in approximately a/Q = .020 and under the same assumptions as for slow crack growth, this results in a specified surface flaw size, 2c = 0.100", a = .050". Obviously for parts where thickness are less than or equal to the specified depths of surface of corner flaws, provisions must be made to handle the analysis of the deep flaw case. The specification stipulates that a through the thickness flaw is assumed for thickness less than the specified depth of cracks (Figure 2.11).

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* 2c = Surface Length

a = critical crack length parameter - depth

Q = shape parameter = F (a/2c) = 2.5 for a/2c = .5

2.4.2 Intact Structure - Marginal Hole Quality

As a means of assessing the quality of fastener holes in military aircraft, regression analyses of crack growth tests have been conducted. The results of these studies indicate that estimates of initial quality for fastener holes can be assigned in terms of an apparent initial flaw and thus degrees of quality can be expressed in terms of the apparent initial flaw size, (i.e., the larger the apparent initial crack, the lesser the quality) (Figure 2.12). The result of these studies indicate that marginal quality holes (i.e., holes containing minor discrepancies of various types) can be characterized as having initial damage equivalent to a small corner crack of the order of .002-.010 inch in radius. Accordingly, MIL-A-83444 assumes that any fastener hole in the structure can be marginal and can have an initial damage equivalent to an .005" radius corner flaw. Thus, it has been assumed that this flaw exists at each fastener hole within the structure at the time of manufacture. Since the .005" size is based on limited data, the contractor may provide data representing his own manufacturing quality and negotiate with the Air Force for a smaller size of apparent initial flaw size to represent marginal hole quality.

The .005" corner flaw representing marginal quality holes is the basis for the fastener policy, continuing damage, and remaining structure damage requirements.

2.4.2.1 Continuing Damage

In applying MIL-A-83444 to a built-up structure, it is observed that cyclic growth behavior of primary damage may be influenced by the geometry of the structure or the arrangement of the elements. The most common influences are: (a) the damage can grow to a free edge and stop, (Figure 2.13) in which case a .005" crack is assumed to be present immediately in order to allow continuation of the growth pattern, (b) the damage can grow and cause an element failure in which case the damage site must be moved to the adjacent fasteners (Figure 2.14) and a new site must be analyzed. In this case, the alternate damage is the .005" crack; however, the assumption is made that it was present initially (i.e., the marginal quality hole). If the new site is the adjacent end fastener of the failed element, then there would be effectively a stepwise shift in the crack growth curve.

2.4.2.2 Fastener Policy

In practice, the growth of flaws from fastener holes can be retarded by the use of interference fit fasteners, special hole preparation (e.g. cold work), and to some degree by joint assembly procedure (e.g. friction due to joint clamp-up). Because of this delayed flaw growth, the slow crack growth lives (or intervals) can be significantly longer than those obtained from structure containing conventional low torque clearance fasteners (Note: In practice it may be possible to permanently delay growth at a flawed hole.)

It is the intent of the fastener policy to encourage contractors to enhance the safety and durability of the structure through the use of these flaw growth retarding fastener/hole preparation systems. Experience has shown that to achieve consistently the beneficial effects of these techniques exceptionally high quality process control is required during manufacture. However, this is not always obtained. As a result, it is thought unwise to consider all interference or hole preparation systems effective in retarding crack growth. On the other hand, there is generally a low probability of having an ineffective interference fastener or no cold work in a hole containing the primary damage (i.e., those specified in Section 3.1.1.1 (a & b) of MIL-A-83444) and it would be unnecessarily conservative to assume this were the case. Accordingly the policy set forth (3.1.1.1c of MIL-A-83444) assumes that any given fastener/hole preparation may be ineffective in retarding flaw growth, however, the assumed initial damage in the hole is equivalent to that associated with a marginal quality hole (.005") rather than the capabilities of non-destructive inspection.

2.4.3 In-Service Inspection Damage Assumptions (Minimum Assumed)

The basic premise in arriving at sizes to assume following an in-service inspection is essentially the same as for the case of intact structure. Once it is established that reliance on in-service inspection is required (as opposed to desired), to insure safety, the initial damage size assumed to exist is that associated with field or depot level NDI capability as opposed to that associated with initial manufacturing inspection capability. However, in special cases where specific part removal at the depot is economically warranted the contractor may recommend that this action be taken. In this case the assumed damage subsequent to part removal and inspection may be smaller and may in fact be the same as in the original design providing the same inspection procedures as used in production are used and certified inspection personnel perform the inspection.

Figure 2.15 summarizes the post inspection damage conditions and/or limitations to which they are applicable. With fasteners installed and sufficient accessibility to the location, the maximum undetectable damage size is 0.25" of uncovered length at fastener holes and, depending upon part thickness, it may be a through or part through flaw. This flaw size was established based on limited available inspection reliability data where the inspection was performed on the assembled aircraft as opposed to the part level inspection performed during production fabrication (Figure 2.9). These assumptions are considered to be applicable for penetrant, magnetic particle, and ultrasonics. Because of lack of sensitivity, X-ray is not considered appropriate for determining tight fatigue cracks and thus is not applicable to these flaw size assumptions.

At locations other than holes or cutouts, a flaw size of surface length 0.50" is assumed to be representative of depot level capability, although this value has not been substantiated by inspection reliability data.

Where visual inspection is performed on the assembled aircraft, the minimum assumed damage is an open through the thickness crack having an uncovered length of 2 inches. This value was established based on visual inspection reliability data derived from inspection of large transport type aircraft during fatigue testing and subsequent teardown inspection (see Figure 2.16).

Note: <u>The data base for establishing values for in-service inspection is limited and in most cases the values are estimates</u>. It is anticipated that current and future planned studies will result in additional data to substantiate or revise the current MIL-A-83444 post inspection flaw sizes.

2.4.4 Demonstration of Flaw Sizes Smaller Than Those Specified for Slow Crack Growth Structure

For the slow crack growth category, an allowance is made for the contractor to select sizes smaller than specified in MIL-A-83444. This may be accomplished by (a) an NDI demonstration program or (b) a proof test:

a. NDI Demonstration Program - As described in paragraph 4.2 of MIL-A-83444, the program must be formulated by the contractor and approved by the Air Force and must verify that for the particular set of production and inspection conditions, flaws will be detected to the 90% probability level with 95% confidence.

b. Proof Test - Proof Test can be an effective means of screening structure for flaws where no other means of NDI is available or where it is indicated to be cost effective. Proof testing generally has been successful for the more brittle materials which follow Plane Strain fracture behavior such as high strength steels. The application of proof testing to complete air frame structure in USAF has been somewhat limited and in general has been used as a last resort to allow operation (usually restricted) until extensive modifications are made to the structure (e.g. B-52D). Proof testing requires the proof stress to be in excess of the maximum operating stress level in order to achieve the maximum benefit. Figure 2.17 illustrates the proof test concept. Since for many materials fracture toughness varies with temperature (higher K_{IC} with higher temperature) and since normal material K_{IC} varies, the sizes of flaws screened out by the proof test inspection must reflect these factors. Therefore the specification requires that the proof test derived minimum initial flaw size be calculated using the upper bound of fracture toughness data (i.e., the larger size, a_1 in Figure 2.17) and the temperature at which the proof test is conducted. Thus, lowering the temperature during the proof test is for some materials a means of reducing the screened flaw size.

2.5 RESIDUAL STRENGTH REQUIREMENTS

2.5.1 General

Required residual strength is defined as the amount of initial static strength available at any time during the service exposure period considering damage present and accounting for the growth of damage as a function of service exposure time. Figure 2.1 (repeated) indicates that strength degrades with increased damage size. The intent of MIL-A-83444 is to provide at least design limit load residual strength capability for intact structure at all times throughout the service life of the structure. The requirement to maintain limit load capability is considered necessary to allow unrestricted operational usage.

The residual strength requirements are specified in terms of the minimum internal member load P_{xx} which must be sustained.

Magnitude of P_{xx} depends upon the service exposure time of the structure between inspections and the overall capability of the inspection. P_{xx} is intended to represent the maximum load the aircraft might encounter during the time interval between inspections. There are other qualifications for P_{xx}. The required P_{xx} is at least design limit load for all intact structure whether or not the structure is being qualified as slow crack growth or fail safe. The required P_{xx} is also at least design limit load when the only planned safety inspections are at the depot (i.e., the depot or base level inspection category).

The goal to allow unrestricted operational capability for all intact structures, has established that for all slow crack growth structure, P_{XX} be, at least, limit load. In addition, all fail safe structure must be designed to be at least depot level inspectable and that P_{XX} over this interval must be at least limit load. For slow crack growth structure this restriction is obvious since the only means to protect the safety is not to allow damage growth to degrade the strength of the structure to less than design limit load. For fail safe structure where partial failure is allowed and subsequent detection of failed load path is required, the restriction on intact structure serves two purposes. First, when coupled with the intact damage growth requirements it provides assurance that, under normal situations, early cracking will not occur (an added durability feature), and second, it is the only way that the operational fleet can be maintained with unrestricted capability. For Fail Safe Multiple Load Path Structure the levels of residual strength must be maintained for the structure at the time of and subsequent to load path failure (see MIL-A-83444, TABLE I).

2.5.2 Residual Strength Requirement for Fail Safe Structure at the Time of Load Path Failure, P_{yy} (Single Load Path Failure Load)

For fail safe structure there is an additional requirement for the remaining structure (at the time of a single load path) to be capable of withstanding at least the load which causes the load path failure, plus an additional increment to account for the dynamic conditions of the breaking member (P_{yy}). While most data and analyses indicate that the dynamic magnification factor associated with the member failure is probably very small, the current specification requires that 1.15 dynamic factor (D.F.) be applied to the amount of load distributed to the remaining structure as the result of a single load path failure.

Since the intact requirements for fail safe structure require that any individual load path be capable of withstanding $P_{LIMIT} \leq P_{xx} \leq 1.2$ $P_{LIFETIME}$, P_{yy} will always be equal to P_{xx} (Intact) times the dynamic factor. Although the specification states that P_{yy} is to be the greater of D.F. times P_{LIMIT} or D.F. times P_{xx} (Intact), the latter will always be the larger because the minimum intact residual strength requirement is at least design limit load.

2.5.3 Determining the Residual Strength Load, P_{xx}, for Fail Safe Structure Subsequent to Load Path Failure

The magnitude of the residual strength load required depends upon the exposure time in service (i.e., the longer the exposure time, the greater the probability of encountering a high load). Accordingly, the value of required P_{xx} load increases with increase in the inspection interval or period of unrepaired service usage (allowable crack growth period). For the short service exposure times between inspections for the In Flight Evident, Ground Evident and Walk Around Visual categories, the probability of encountering limit load conditions is low and thus the required P_{xx} may be significantly below design limit load. For the longer exposure times this is not the case and, as stated previously, the minimum required P_{xx} for structure qualified under the non-inspectable or depot level inspectable categories must be at least limit load.

The value of P_{xx} is established from load spectra data derived for a mission analysis of the particular aircraft considering average usage within each mission segment. Unless otherwise stated, MIL-A-008866B is the basic source of load factor data for the various classes of aircraft. Since safe operation depends upon the residual strength capability and because any individual fleet aircraft may encounter loads in excess of the average during the particular exposure time, the required P_{xx} load should be larger than the average derived value. One way to accomplish this is to magnify the inspection interval by a factor M (e.g., increase the service exposure time for the aircraft between inspections). This is the method used in MIL-A-83444. The values of M, as specified in Table I of MIL-A-83444, are summarized in Table 2-2. For example, under the depot level inspectability category, the P_{xx} load is the maximum value expected to occur in 20 times a typical inspection interval.

The basis for the specified M values is somewhat arbitrary although it is felt that the loads derived by this method are not unreasonably conservative. The basis for M = 100 is exceedance data for transport type

TABLE 2-2 Inspection Interval Magnification Factors

P_{XX}*	Degree of Inspectability	Typical Inspection Interval	Magnification Factor, M
P_{FE}	In-Flight Evident	One Flight	100
P_{GE}	Ground Evident	One Flight	100
P_{WV}	Walk-Around Visual	Ten Flights	100
P_{SV}	Special Visual	One Year	50
P_{DM}	Depot or Base Level	1/4 Lifetime	20
P_{LT}	Non-Inspectable	One Lifetime	20

*P_{XX} = Maximum average internal member load that will occur once in M times the inspection interval. Where P_{DM} or P_{LT} is determined to be less than the design limit load, the design limit load shall be the required residual strength load level. P_{XX} need not be greater than 1.2 times the maximum load in one lifetime if greater than design limit load.

aircraft where it has been observed that shifting exceedances by approximately two decades (i.e., M = 100) magnifies the value of load factor by approximately 1.5 (Figure 2.18). It was recognized that for fighter data, exceedances approaching or exceeding design limit values are probable but that extrapolation of the basic exceedance curve very far beyond limit n_z is often meaningless and unwarranted due to physical limitations of the vehicle and crew. Furthermore, in most cases actual service data is somewhat sparse for this region of the curve. Therefore, it was recognized that (1) an upper limit was required on P_{xx} for fighter aircraft and (2) the value of M should be less for longer inspection intervals in order that unreasonable factors would not be imposed should the actual derived P_{xx} be less than the specified upper limit. The values of M = 20, M = 50 are arbitrary but probably not unreasonable (see Figure 2.18). Where P_{xx} is derived to be in excess of that associated with the design limit conditions, P_{xx} need not be greater than 1.2 times the maximum load expected to occur in one design lifetime.

The procedure for obtaining P_{xx} is illustrated in the following example: (Figure 2.19)

- Consider average exceedance data for one design lifetime
- Max load expected in one lifetime is in excess of limit load (Point A)

- Shifting curve A to B and extrapolation to C to represent a twenty lifetime exceedance curve, yields P_{xx} (derived) at C. P_{xx} then is either the value derived at C or 1.2 x (load amount at A) which even is smaller. In this case $P_{xx} = P_{LT}$ is the load at point C.

2.6 REQUIRED PERIODS OF SAFE DAMAGE GROWTH (PERIOD OF UNREPAIRED SERVICE USAGE

The required periods of safe damage growth are specified in terms of either the design service lifetime or the scheduled inspection interval. Various factors have been applied to these usage periods as described below.

2.6.1 Slow Crack Growth Non-Inspectable Structure

The required period is two times the design lifetime. A factor of two is applied to cover various uncertainties associated with crack growth during service usage that may not be adequately accounted for in analyses or laboratory test.

2.6.2 Slow Crack Growth Depot Level Inspectable Structure

The required period is two times the depot level inspection interval. A factor of two is applied to allow for one missed inspection and still enable flaw detection and repair prior to failure.

2.6.3 Fail Safe Structure - Intact Requirements

The required period is one design lifetime or one depot level inspection interval. As previously mentioned, these requirements are not for safety, specifically, but have been imposed to help prevent adverse durability problems in multiple load path construction which could jeopardize unrestricted operational capability of the aircraft. A factor of one appears appropriate since safety is not involved and because separate durability requirements (as contained in MIL-A-008866B) must be met by all structures.

2.6.4 Remaining Structure - Fail Safe Categories

The period (referred to as the "period of unrepaired service usage") depends upon the inspectability level.

For structure where the damage is classified as In Flight Evident this period is the time required to return to base. For structure classified as Ground Evident this period is a single flight. For these two cases a factor of one is applied. This is justified on the basis that in order for the structure to be categorized in these inspectability levels, damage detection must be a certainty. For Walk Around Visual inspections detection of failed load paths, arrested cracks and or large subcritical cracks is not a certainty during any single inspection. Accordingly, an arbitrary factor of 5 is applied to the inspection interval. For Special Visual this factor is reduced to 2 because of the more detailed nature of such inspections and the resulting improved confidence in detection. The specified periods are contained in paragraph 3.2.2.2.2 of MIL-A-83444 and are repeated in TABLE 2-3.

TABLE 2-3 Required Periods of Safe Crack Growth - Remaining Structure Fail Safe Categories

Degree of Inspectability	Minimum Period of Unrepaired Service Usage
In-Flight Evident	Return to base
Ground Evident	One Flight
Walk-Around Visual	5 x Inspection Interval - 5 x 10 Flights
Special Visual	2 x Inspection Interval - 2 x One Year
Depot or Base Level	2 x Inspection Interval - 2 x One Quarter Lifetimes

2.7 ILLUSTRATIVE EXAMPLE OF THE APPLICATION OF MIL-A-83444

2.7.1 Structural Design

The example chosen is representative of a lower wing structure and is comprised of multiple skin and stringer elements (Figure 2-20). The skin panels 1-5 are considered the major load paths. At each spanwise splice a major splicing stringer is located and the construction is such that the load paths are independent, that is no common manufacturing tie exists between the skin panels.

2.7.2 Design Service Life - Assume the design service life is 40,000 hours.

2.7.3 Choice of Structural Design Concept

In the initial example the structure will be considered as fail safe multiple load path and the steps required to satisfy this requirement will be outlined. Later the same problem will be examined as a slow crack growth qualified design. The structure will be designed to be fail safe by virtue of being able to sustain the failure of one major load path or skin panel and still maintain the residual strength and remaining structural requirements. For illustration purposes the critical load path will be chosen as panel #2. Although 2 is critical from a remaining structure point of view, every panel must be designed to meet the intact requirements.

2.7.4 In Service Inspection Considerations

Since the design is intended to satisfy the fail safe multiple load path category, an in-service inspection plan is required. Assume that the lower surface will be periodically inspected in the field by a walk around visual type examination, generally unaided. The frequency of these inspections is approximately every ten flights. In addition, the structure will undergo a depot level inspection at approximately 1/4 design lifetime intervals or every 10,000 hours. During manufacture, conventional inspection methods will be conducted and a fracture control program will be instituted.

2.7.5 Initial Flaw Considerations

Flaws assumed to result from manufacturing and/or material conditions are specified in 3.1.1.1 of MIL-A-83444 for fail safe structure. The primary damage at a fastener hole (Figure 2-21) is an .020" corner flaw and since the drilling operation is common to the skin and splicing stringer, the .020" flaw must be assumed in both members. Since panel 2 is considered as critical (i.e., the minimum residual strength occurs with 2 failed) panel 2 will be considered in this example.* Note that only one primary damage site is assumed for each load path (e.g. along the path or growth of the damage, along a wing station). Also, it is not necessary to consider the interaction of

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* The intact structure requirements must be checked for each major load path independently. Only 2 is considered here.

flaws from adjacent primary sites. Each analysis of primary damage is conducted independently. At each hole other than the assumed primary site, an .005" radius corner flaw is assumed to represent average or typical manufacturing quality. The interaction of the .005" flaws with the primary flaws must be considered when conducting the analysis.

2.7.6 In-Service flaw assumptions following inspection

The capability of inspection in the field is generally less than at the depot. The sizes of damage assumed to exist following inspection are specified in 3.1.2 of MIL-A-83444. For this example, assume that penetrant or ultrasonics will be used at the depot both exterior and interior to the lower surface. If this type of inspection is conducted, the damage likely to be found will be much smaller than the failed skin panel. From 3.1.2 (b) of MIL-A-83444 the minimum damage size to be assumed is a through crack of 0.25" uncovered length. The locations of the 0.25" length both in the skin and in the splicing stringer should be selected on the basis of inspectability but should be the location most critical to subsequent growth. Assume for purposes of illustration, that the damage is as indicated in Figure 2.22. This figure also illustrates that .005" continuing damage (3.1.1.2a) is required to complete the flaw growth analysis for this damage condition. The .005" flaw away from the primary damage site represents the initial manufacturing type damage as specified in Para. 3.1.1 of MIL-A-83444.

2.7.7 Remaining Structure Damage following the failure of the Major Load Path

Figure 2.23 illustrates the condition of the structure following the complete failure of the primary load path (skin panel 2), represented by W. The condition of the remaining structure is as specified in 3.1.12 (b) of MIL-A-83444 since this is an example of independent structure. Each fastener hole in the structure is assumed to contain the .005" typical manufacturing hole quality flaw. The Δa_2 increment is the growth of these typical flaws from the time of manufacture until the point at which the load path is assumed to have failed. The increment Δa_2 will be discussed later.

2.7.8 Analysis of Intact Structure - Residual Strength Requirements and Damage Growth Limits (3.2.2.2.1)

The specific set of requirements for intact structure depends upon the capability of the depot level inspection. Since this example has illustrated the situation where the normal inspection can detect less than a failed load path, this case will be examined first:

The intact requirement is that the size damage assumed to be present following the depot level inspection (Fig. 2-22) shall not grow and cause failure of the major load path (i.e. panel 2) before the next opportunity to discover the damage (i.e., the next inspection).

Since this is merely a one time design requirement not specifically intended for safety, it is not necessary to account for the time at which the requirement is imposed (i.e., the structure is considered as "new" and no incremental growth Δa is computed). Figure 2.24 illustrates schematically the residual strength and growth requirements that must be met for the intact structure.

2.7.9 Analysis of Intact Structure (Alternate Requirement)

If it were determined that the depot level inspection was incapable of finding damage less than a failed load path, then the requirement for intact structure is:

Initial manufacturing damage (3.1.1.1) shall not grow to the size required to cause load path failure due to the application of P_{LT} in one design lifetime. The initial damage assumption for this case is illustrated in Figure 2.21. The schematic of the growth and residual strength requirements are illustrated in Figure 2.25.

2.7.10 Discussion of Intact Structure Analysis

Although the structure in the example was assumed to be level inspectable for less than a failed load path, the intact structure requirement associated with this category data might have been more difficult to meet than if the structure had not been inspectable for less than a failed load path. If this were the case it would be satisfactory to qualify this structure under the alternate requirement (Figure 2.25). As is often the case, the designer may choose to qualify the structure in the easiest (analysis) manner providing no undue penalty (e.g. weight) is placed on the design.

2.7.11 Analysis of Remaining Structure Subsequent to Load Path Failure

The fail safe characteristics of this structure (i.e., the ability to fail panel #2 and fly safely until the failed panel is detected) depends upon the residual strength capability at the time of and subsequent to load path failure and the capability of and frequency of in-service inspections. The remaining structure requirements are specified in 3.2.2.2.2 of MIL-A-83444. For this example, the fail safety will be supported by walk around visual inspections for damage sizes of the order of a failed load path. Generally, the walk around visual inspection can be aided by such detectability factors as signs of fuel leakage. At any rate, the minimum inspection capability for this example will be considered to be a failed load path.

Thus, the damage as illustrated in Figure 2.23 shall not grow to a size such as to cause loss of the wing due to the application of P_{sv} in 5 times the inspection interval, or 5 x 10 = 50 flight. This is illustrated in Figure 2.26. Note that $P_{xx} = P_{wv}$ will generally be less than the design limit condition and P_{yy} as discussed in Section 2.5 will always be equal to or greater than that associated with the design limit condition.

2.7.12 Derivation of Residual Strength Load P_{yy}

In the analysis of the intact structure, the critical damage limit was failure of the skin panel 2. The mode of failure was slow growth of either initial manufacturing damage or depot level inspection type damage (Figure 2.24 and 2.25). In each case the damage is assumed to grow in a stable manner until the critical damage size in the skin panel is reached. The critical damage size for this case would be that size at:

$$P_{LIMIT} \leq P_{xx} \leq 1.2\ P_{ONE\ LIFETIME}$$

For a balanced fail safe design, the remaining structure must be capable of withstanding the effects of the major load path failing, including the redistribution of load to adjacent members at the time of load path failure. This is the basis for the requirement that the remaining structure must support the P_{yy} residual strength load. P_{yy} is dependent upon the design allowable for the first panel (Panel 2 in this case). Assume for example that P_{xx} allowable for first panel failure is exactly P_{LIMIT}. The remaining structure must be capable of supporting P_{LIMIT}, with adjacent panels carrying the increment or that portion originally carried in panel 2 at P_{LIMIT}. This is illustrated in Figure 2.27. In Figure 2.28, the amount of load in panel 2 at the limit design condition is redistributed as $(\Delta P_1 + \Delta P_3 + \Delta P_4)$. This increment, P_2 is multiplied by 1.15 at account for dynamic effects. The total redistributed increment then is

$$1.15\ P_2 = (\Delta P_1 + \Delta P_3 + \Delta P_4)$$

The residual strength of the remaining structure is then checked against this conditon.

2.7.13 Incremental Damage Growth Δa

The remaining structure analysis of damage growth and residual strength considers damage in the adjacent structure at the time of load path failure which has grown an amount Δa from the time of manufacture (Figure 2.25). Since the structure must meet the single design lifetime requirement, it becomes necessary to establish at what point during the lifetime the failure of the load path is assumed to take place so that the proper amount of growth Δa can be computed to represent growth during this time segment. Figure 2.29 illustrates the growth of the .005" manufacturing type damage from time zero for one design lifetime. In this example the walk around visual inspection is being called on to detect the failure of the major load path and the inspection interval is 10 flights. MIL-A-83444 requires a factor of 5 on this interval and thus the damage growth life requirement is 50 flights. Therefore, the maximum amount of Δa and the condition to be met would be growth for one design lifetime minus 50 flights. For any other in-service inspection interval the amount Δa would be computed in a similar manner. For example, if the walk around visual inspection was not conducted and fail safety was dependent upon discovery of damage at the scheduled 10,000 hour depot level inspection, then the increment of growth Δa_2 would be one design lifetime minus 2X (10,000 hrs.) as in Figure 2.30.

2.7.14 Alternative-Analysis of Remaining Structure Subsequent to Load Path Failure

As indicated in 2.7.13, the designer may choose to depend upon the depot level inspection instead of the walk around visual. This would be a satisfactory alternative and for this situation the assumption would be made that the major load path failed between depot level inspections and that the aircraft would be designed to operate safely with the failed load path until the next depot inspection. Figure 2.31 illustrates this case.

2.7.15 Qualification as Slow Crack Growth

The previous example illustrated the steps required to qualify the structure under the category of multiple load path fail safe. For that category, an intact requirement (prior to load path failure); a residual strength requirement at the time of load path failure and a remaining structure damage growth and residual strength requirement had to be met. Generally, this is a complex set of analyses to make and in the early design stage may be impractical. The design could be made to satisfy slow crack growth requirements, either non-inspectable or depot level inspectable, while still maintaining some level of ail safety (but not necessarily meeting the requirements specifically). This approach would generally be satisfactory and usually requires a lesser amount of analysis, particularly for computing residual strength and the growth increment.

2.7.15.1 Slow Crack Growth-Non In-Service Inspectable

For this case no special in-service and no depot level inspections will be specifically required to protect the safety of the wing structure. The specific requirements are described in 3.2.1.2 of MIL-A-83444.

2.7.15.1.1 Initial Flaw Sizes Assumed to Result from Manufacturing

Flaws assumed are specified in 3.1.1.1 of MIL-A-83444 for the slow crack growth type structure. In the example chosen, this is an .050" corner flaw at the critical fastener hole joining panel 2 and splicing stringer (Ref. Figure 2.32).

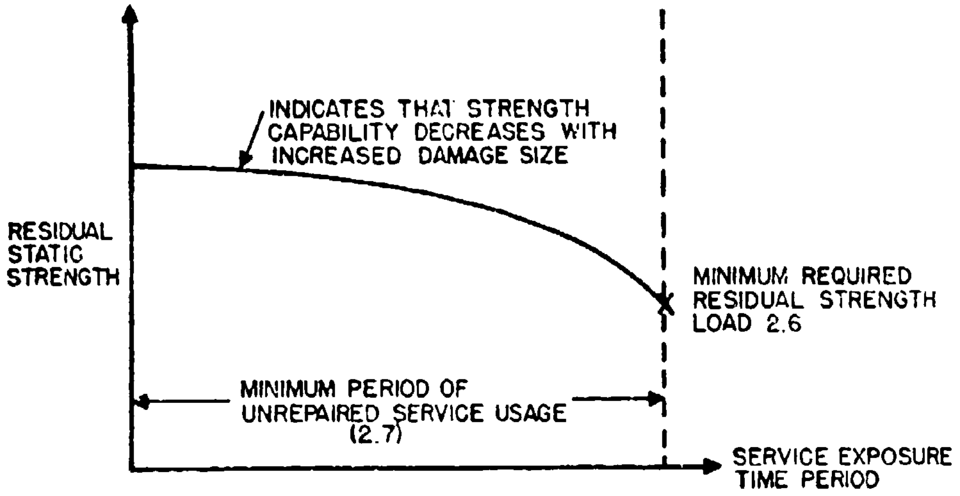
2.7.15.1.2 Residual Strength Load, P_{XX}

The required level of residual strength for non-inspectable structure is P_{LT}, the maximum load expected to occur in one lifetime.

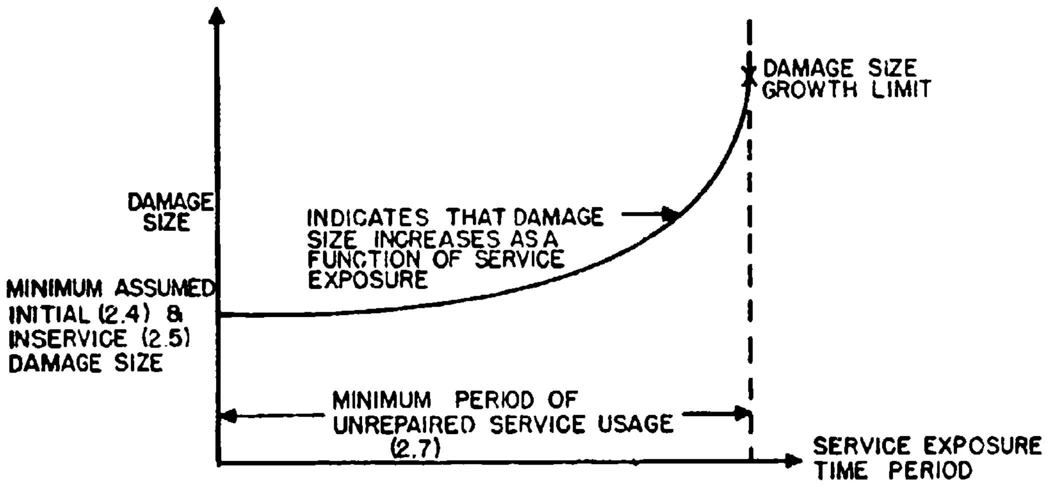
2.7.15.1.3 Analysis Requirements - The slow crack growth and residual strength requirements for this category are illustrated in Figure 2.33. Note that the damage limit in the ultimate is failure of the wing. Engineering judgement may dictate that a more reasonable limit and, perhaps, an easier situation to adhere to would be to establish the limit at some intermediate point such as the failure of one primary load path (W). This might be accomplished in design at very little expense to overall weight.

2.7.16 Slow Crack Growth - Depot or Base Level Inspectable

For this case, the planned 1/4 lifetime or 10,000 hour depot level inspection interval would be relied upon to detect sub-critical damage following this inspection with the provision that the starting initial flaw size would be just smaller than the established depot or base level capability. The assumed size is specified in 3.1.2 of MIL-A-83444 and for this example is identical to that assumed for the intact portion of the fail safe category (Figure 2.22). The required residual strength and damage growth limits are specified in 3.2.11 of MIL-A-83444, and illustrated in Figure 2.34.



(a) RESIDUAL STRENGTH DIAGRAM



(b) DAMAGE GROWTH DIAGRAM

Figure 2.1 Residual-Strength and Damage-Growth Requirements

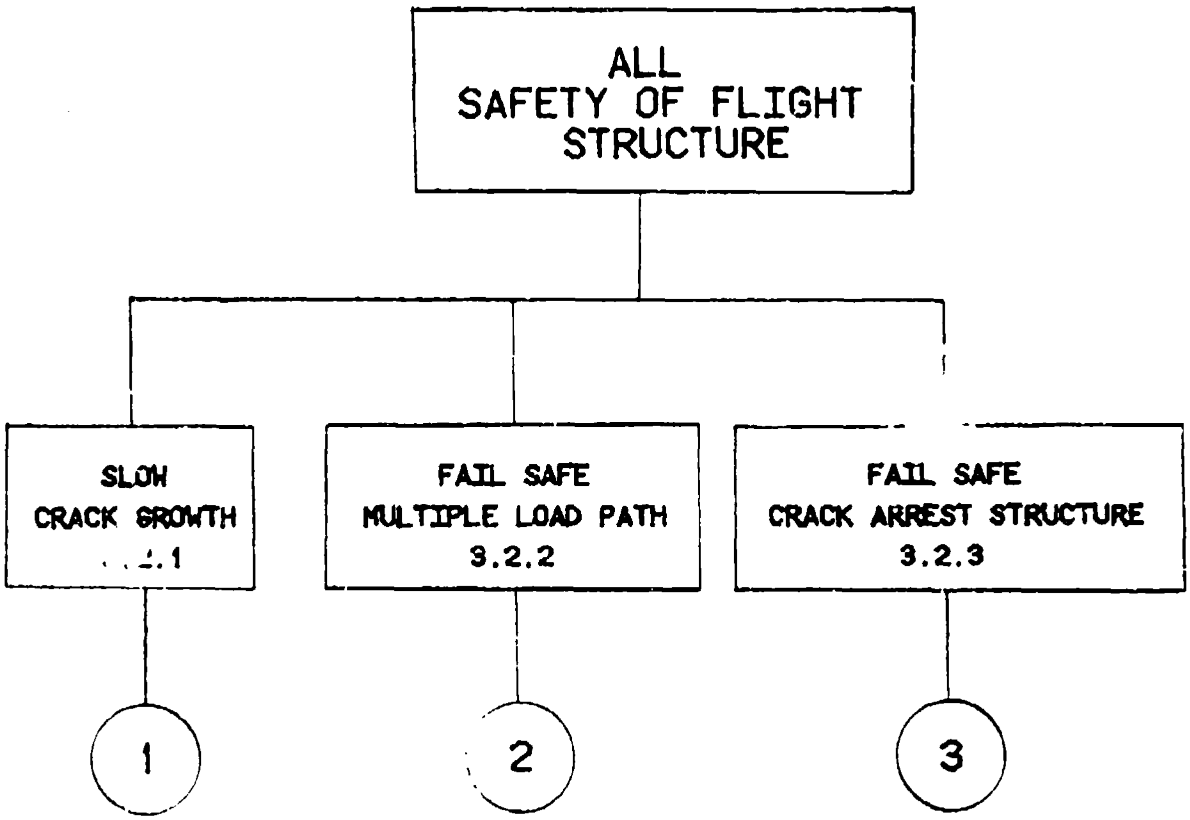


Figure 2.2 Essential Elements of MIL-A-83444

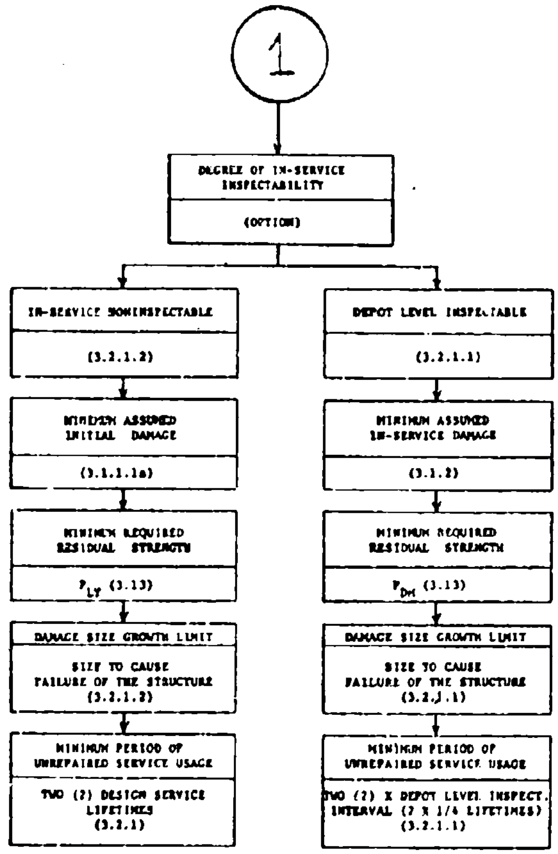


Figure 2.2 (Con't)

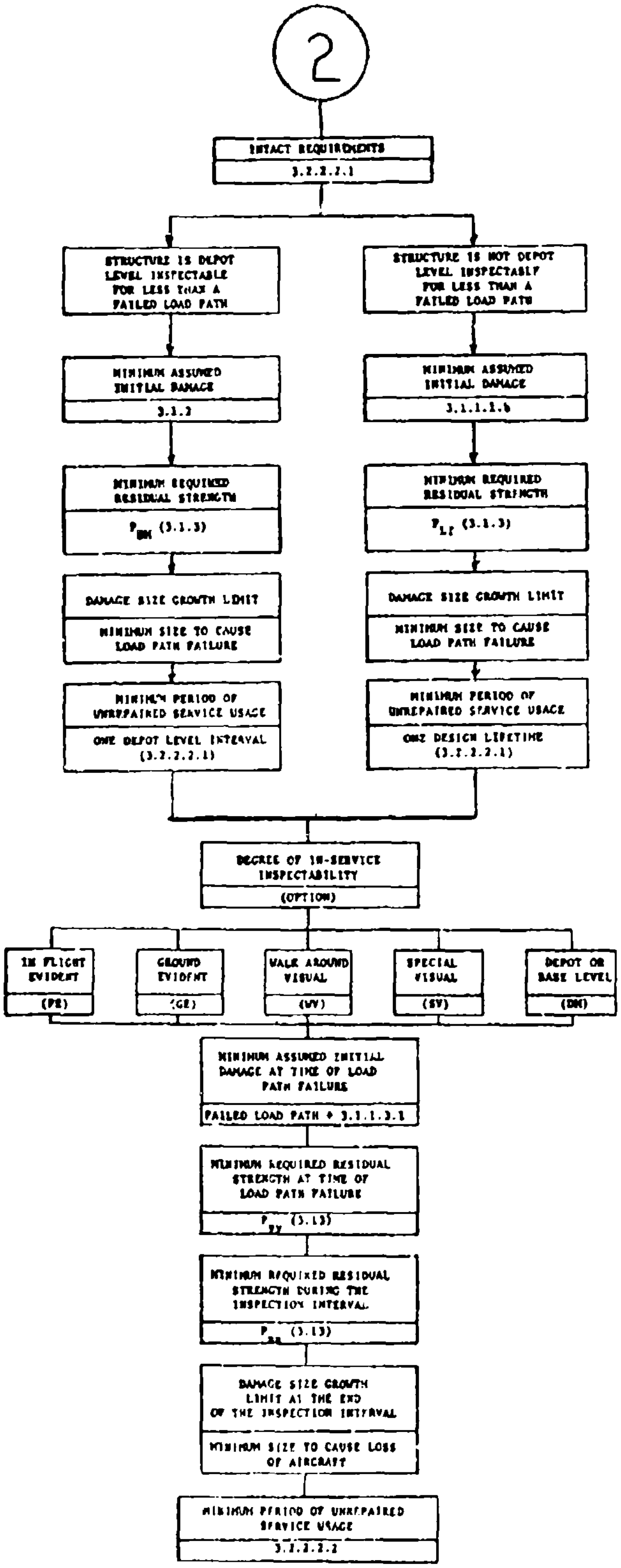


Figure 2.2 (Con't)

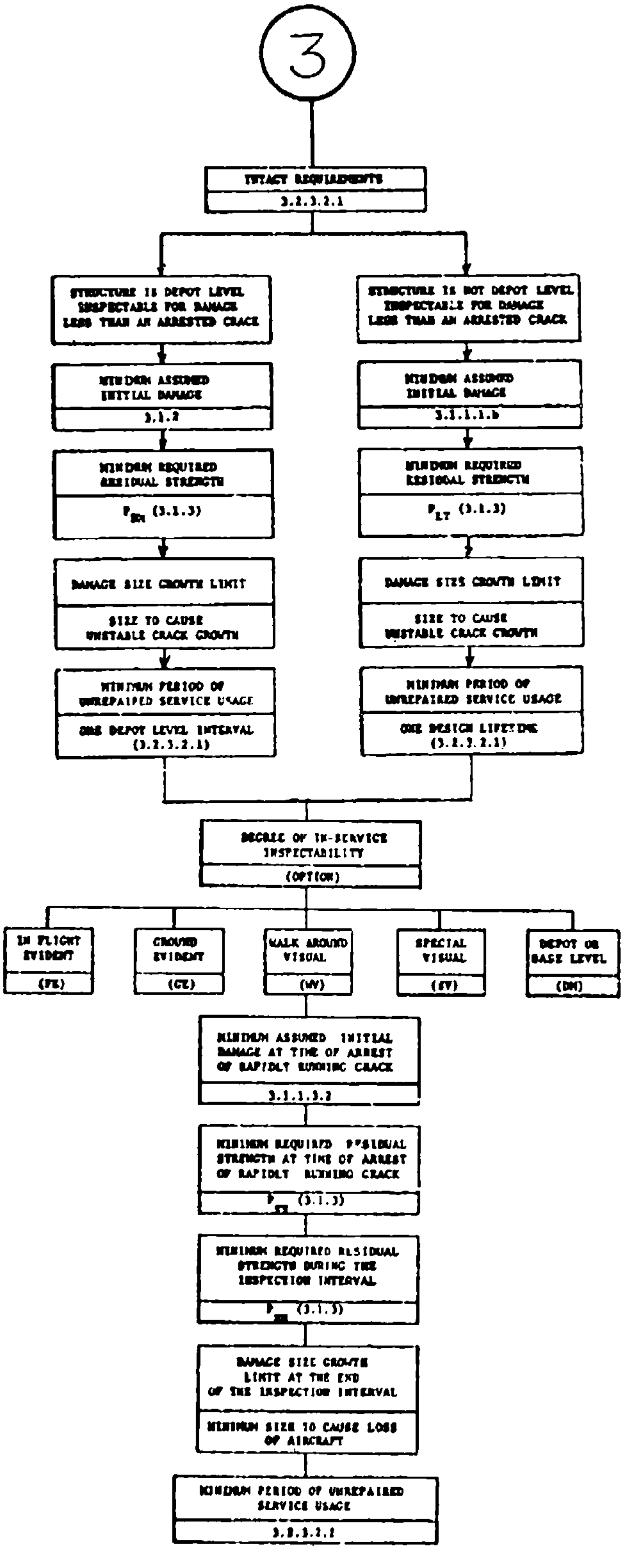
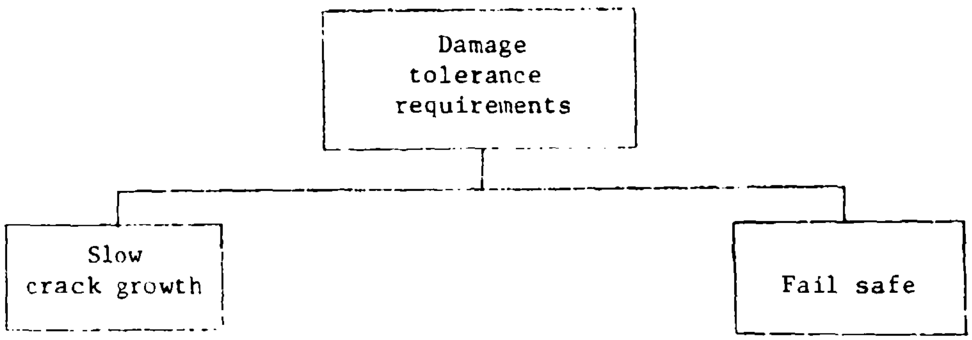


Figure 2.2 (Concluded)



This category includes all types of structures, single and multiple load path which are designed such that initial damage will grow at a stable, slow rate and not achieve a size large enough to fail the structure for a specified slow crack growth period. Safety is assured by the slow rate of growth.

Usually structure comprised of multiple elements or load paths such that damage can be safely contained by failing a load path or by the arrestment of a rapidly funning crack at a tear strap or other deliberate design feature. Fail safe structure must meet specific residual strength requirements following the failure of the load path or the arrestment of a running crack, safety is assured by the allowance of a partial failure of the structure, the residual strength and a period of usage during which the partial failure will be found.

Figure 2.3 Damage-Tolerance Structural Design Categories

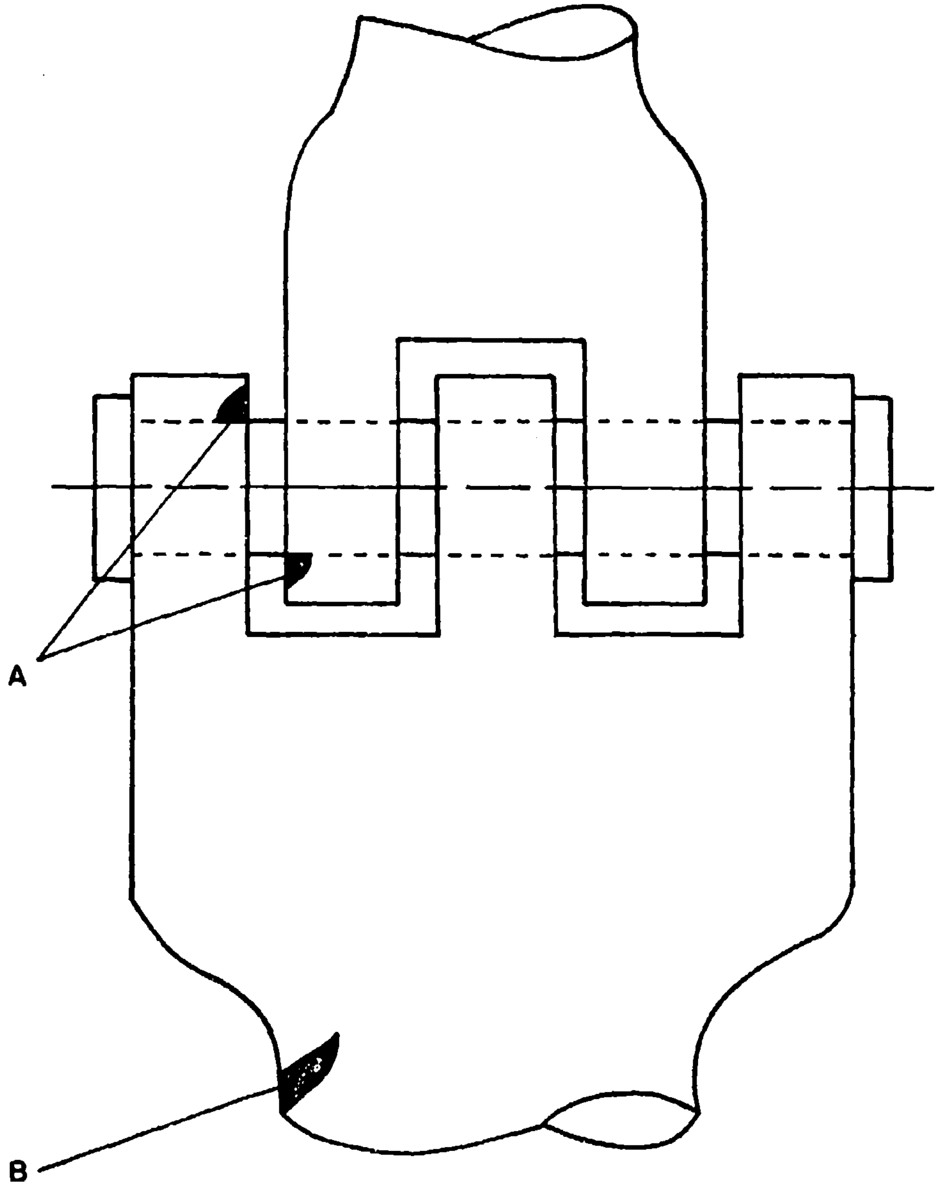


Figure 2.4 Lug Example

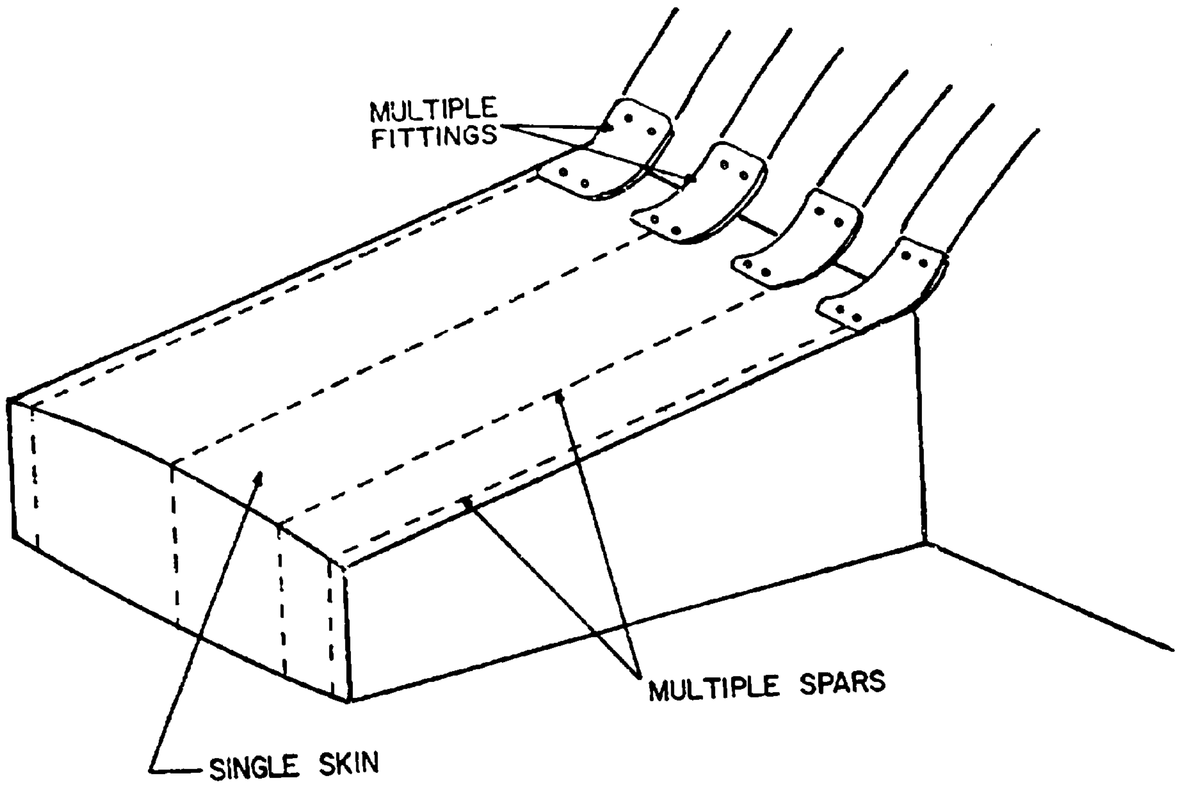


Figure 2.5 Wing Box Example

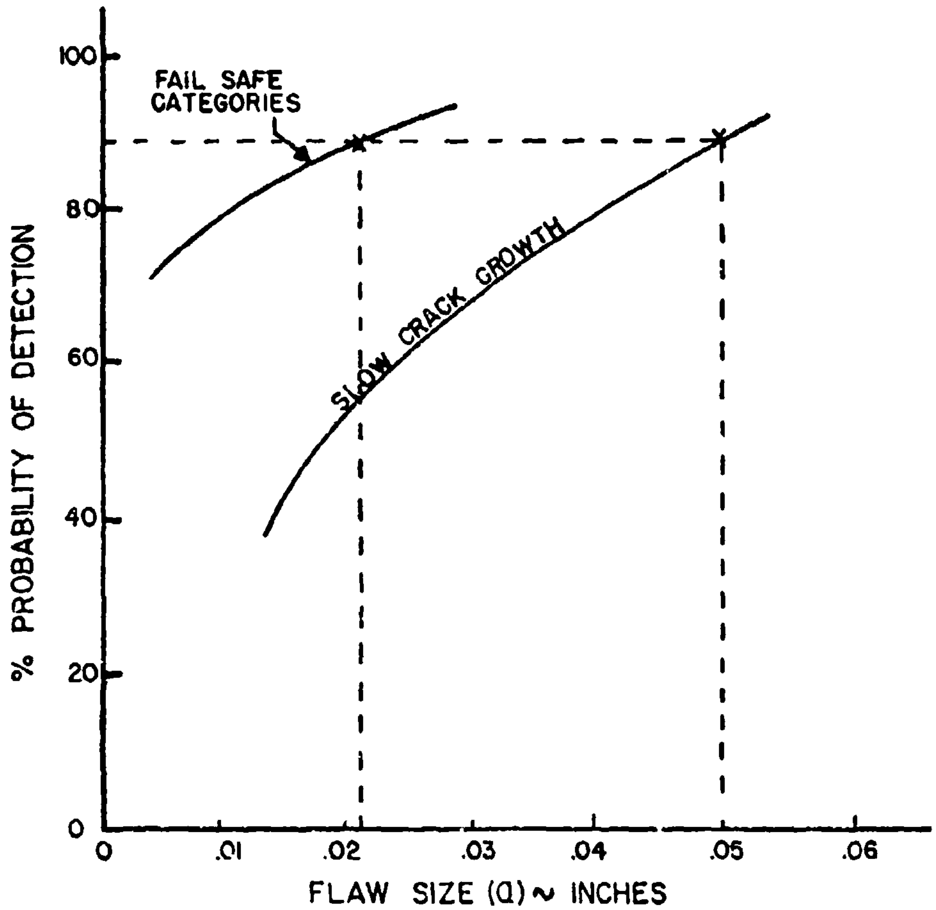
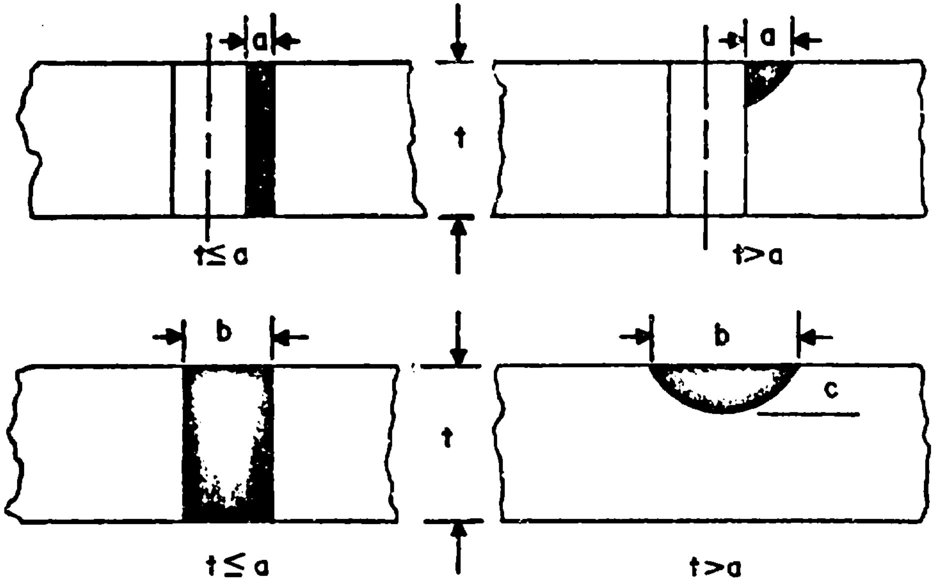


Figure 2.6 Schematic Representation of Rationale for Selecting Initial-Damage Sizes



Locations other* than holes

PRIMARY DAMAGE (in)*		
	SLOW CRACK GROWTH	FAIL SAFE
a	0.050	0.020
b	0.250	0.100
c	0.125	0.050

* OTHER POSSIBLE SHAPES SHALL BE CONSIDERED WITH EQUIVALENT STRESS INTENSITY K

Figure 2.7 Summary of Initial-Flaw Assumptions for Intact Structure

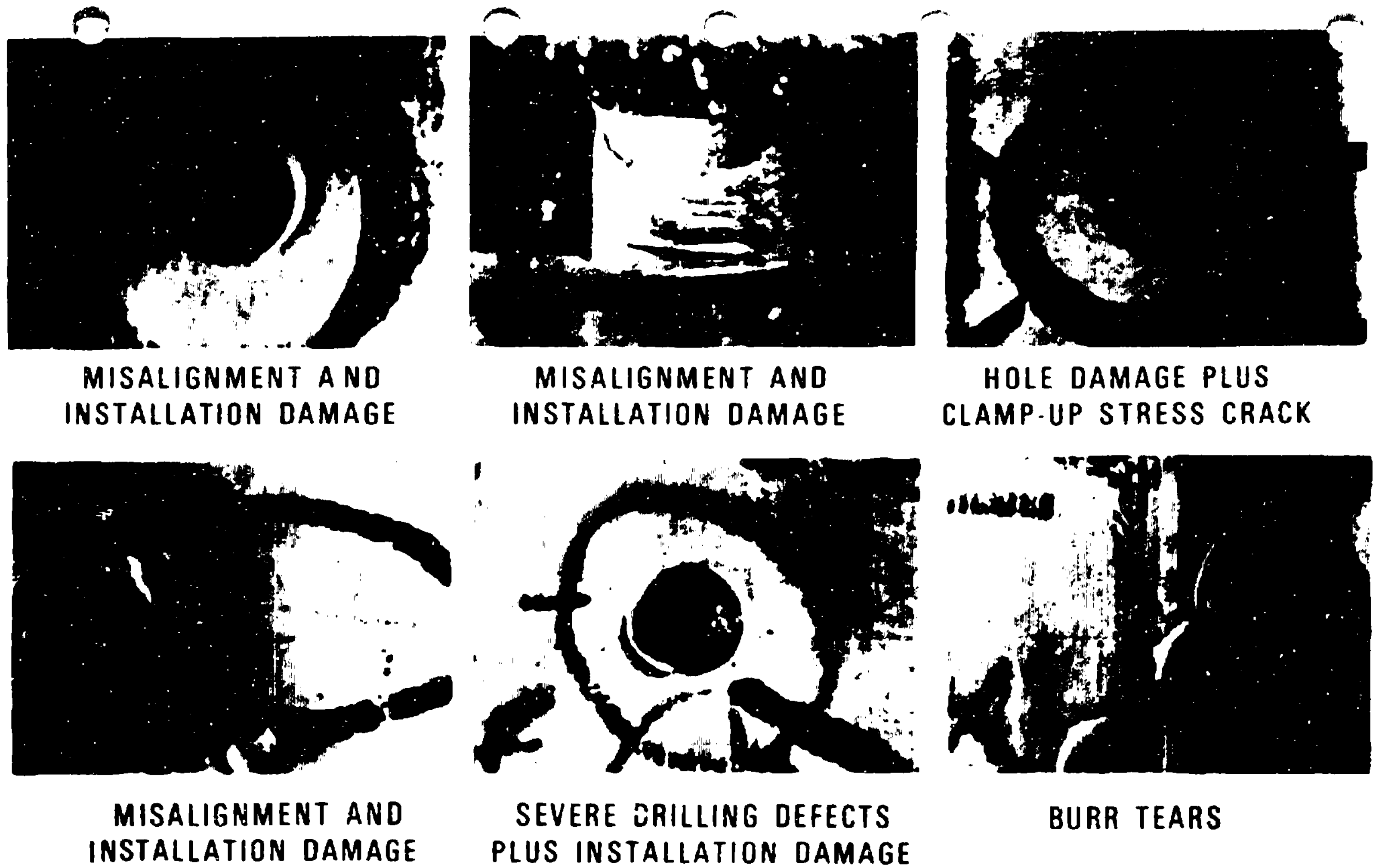


Figure 2.8 Examples of Fastener Holes - Preparation and Assembly Damage

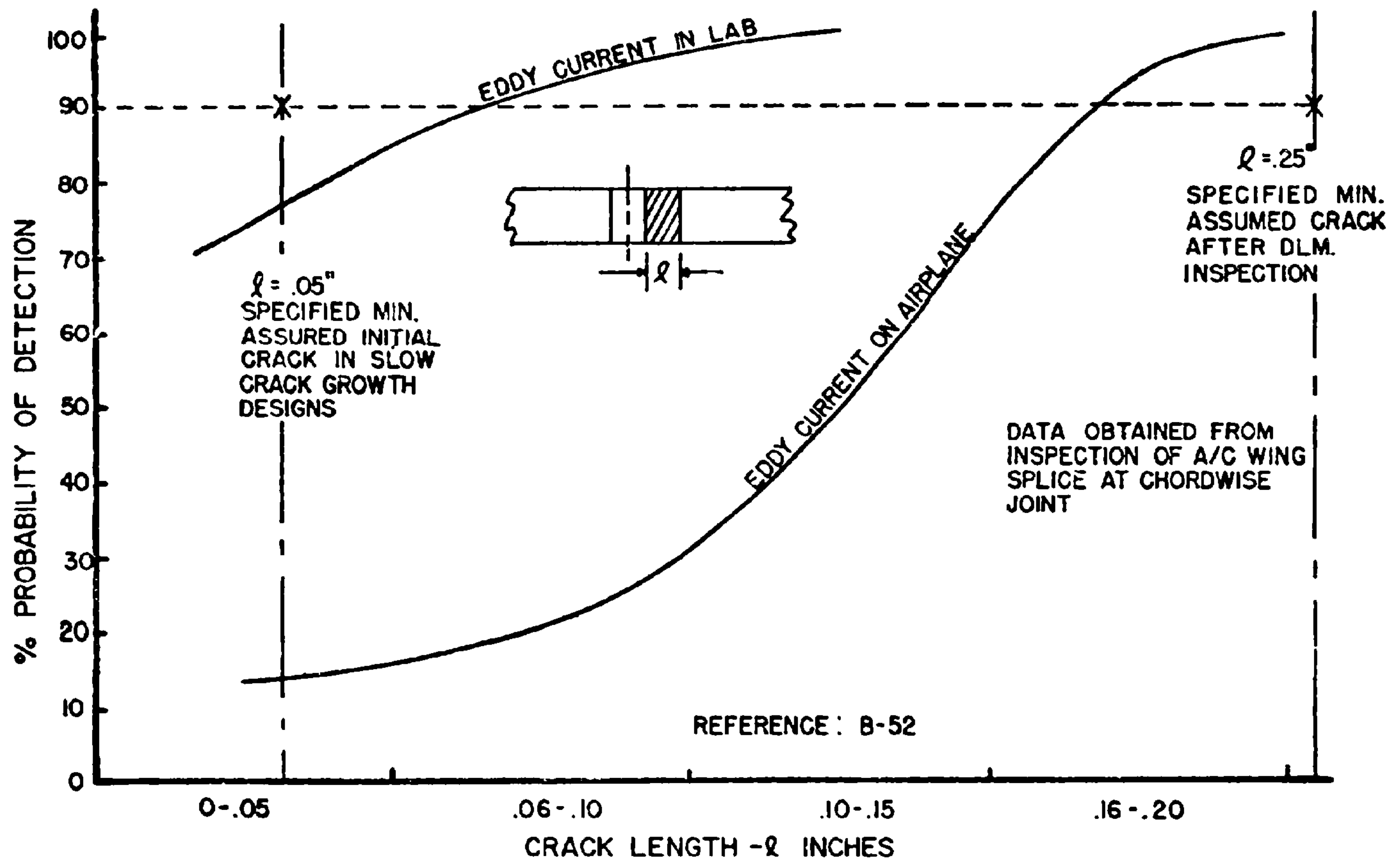


Figure 2.9 Reliability of Eddy Current Inspection for Detecting Cracks at Fastener Holes

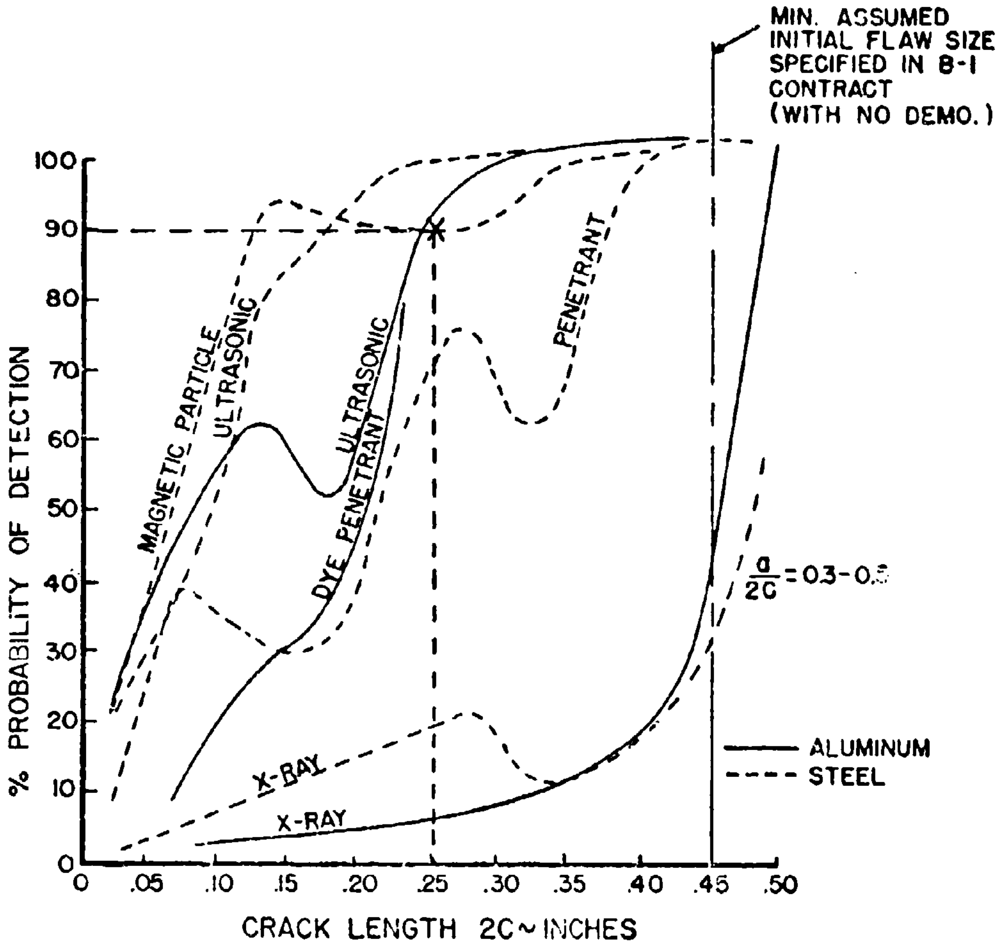


Figure 2.10 Reliability of Manufacturing Inspection for Detecting Surface Flaws

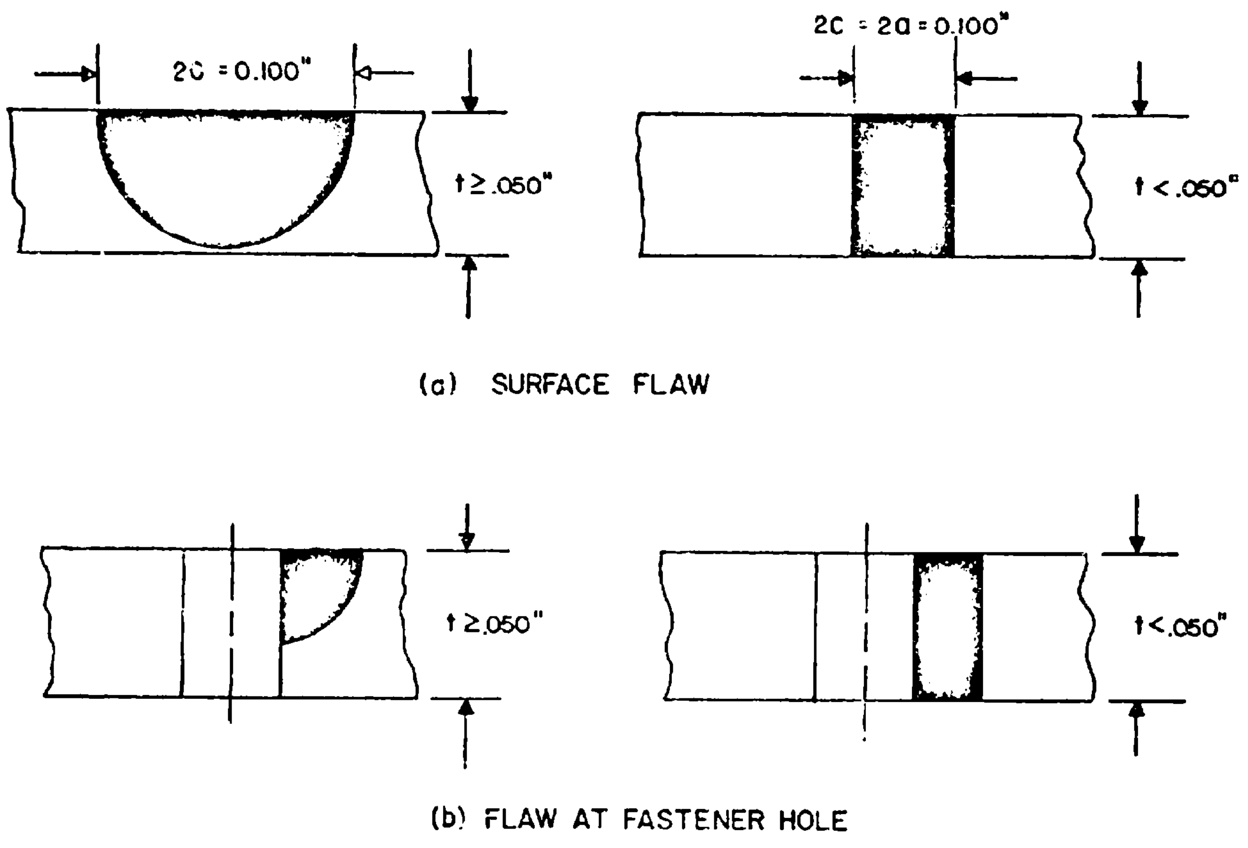


Figure 2.11 Illustration of Thickness Criteria for Assuming Initial Flaws

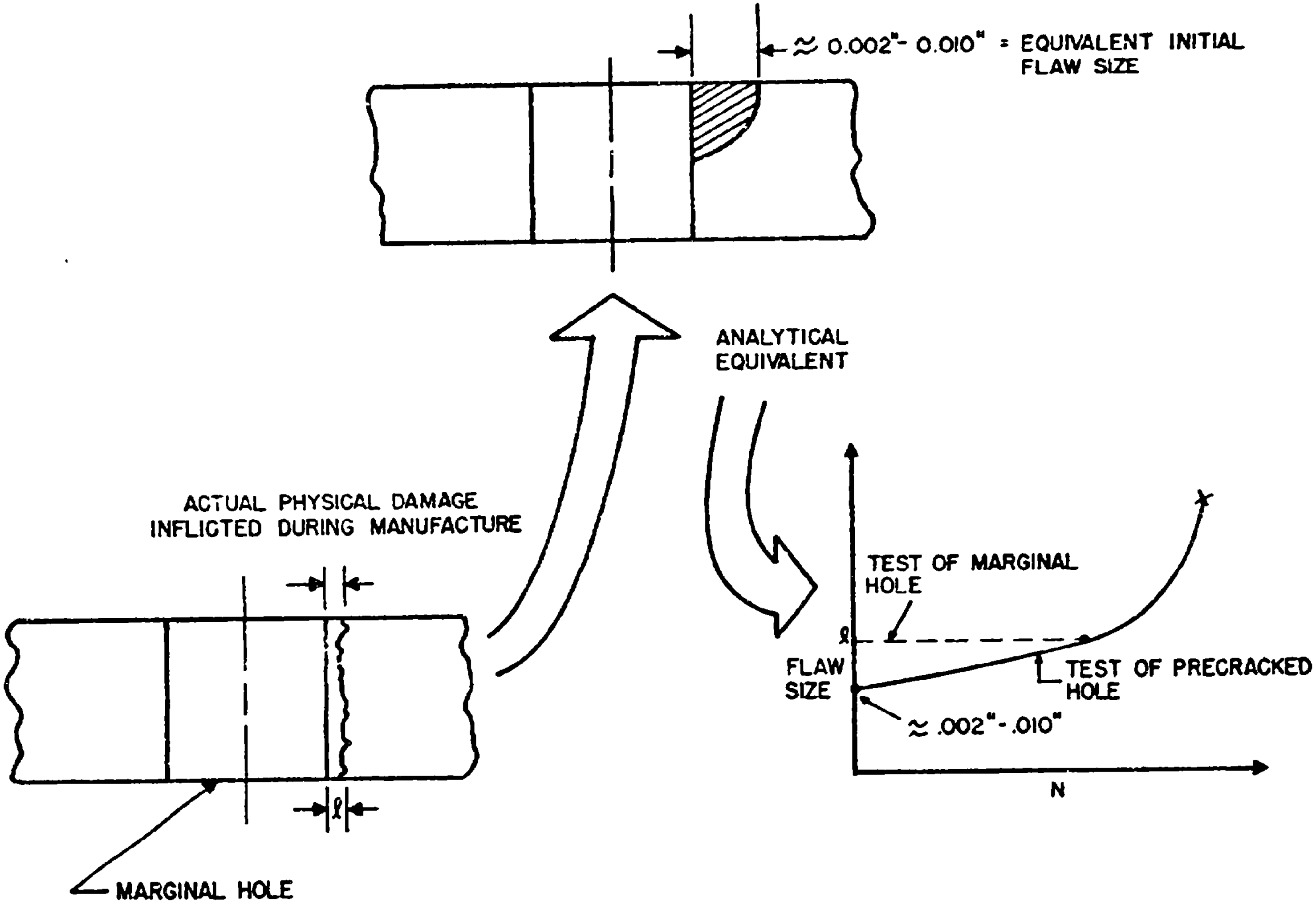


Figure 2.12 Representation of Marginal Hole Quality

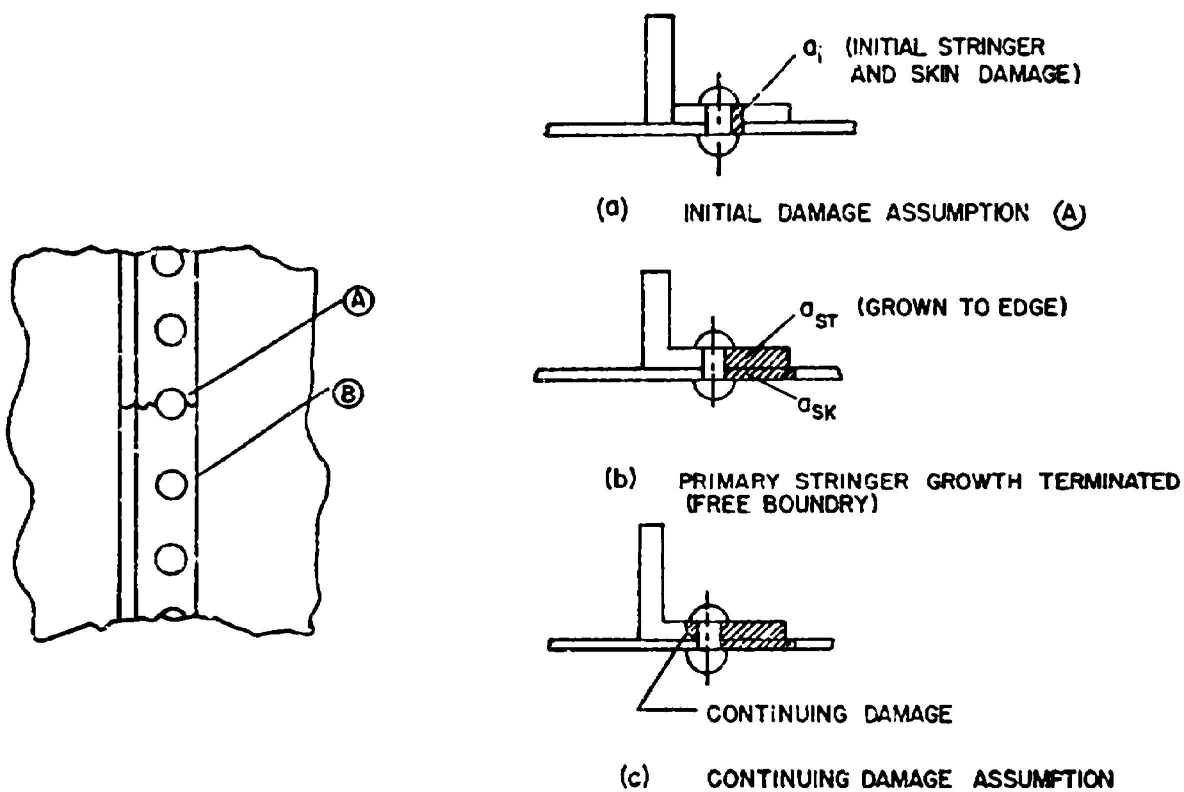


Figure 2.: ample of Continuing Damage
ow-Crack-Growth Structure
Growth of Damage Terminated
at Free Edge

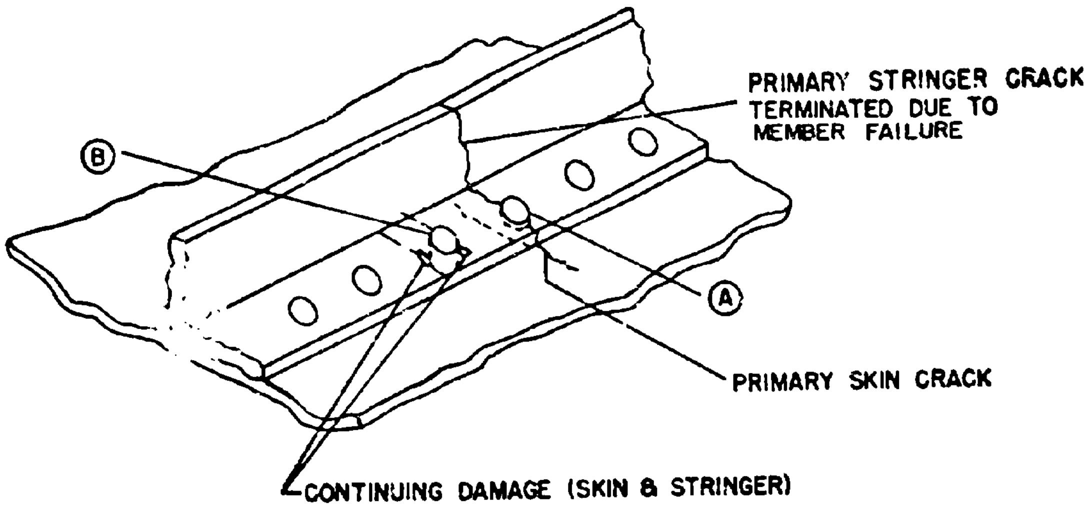


Figure 2.14 Example of Continuing Damage Slow-Crack-Growth Structure Growth of Primary Damage Terminated Due to Element Failure

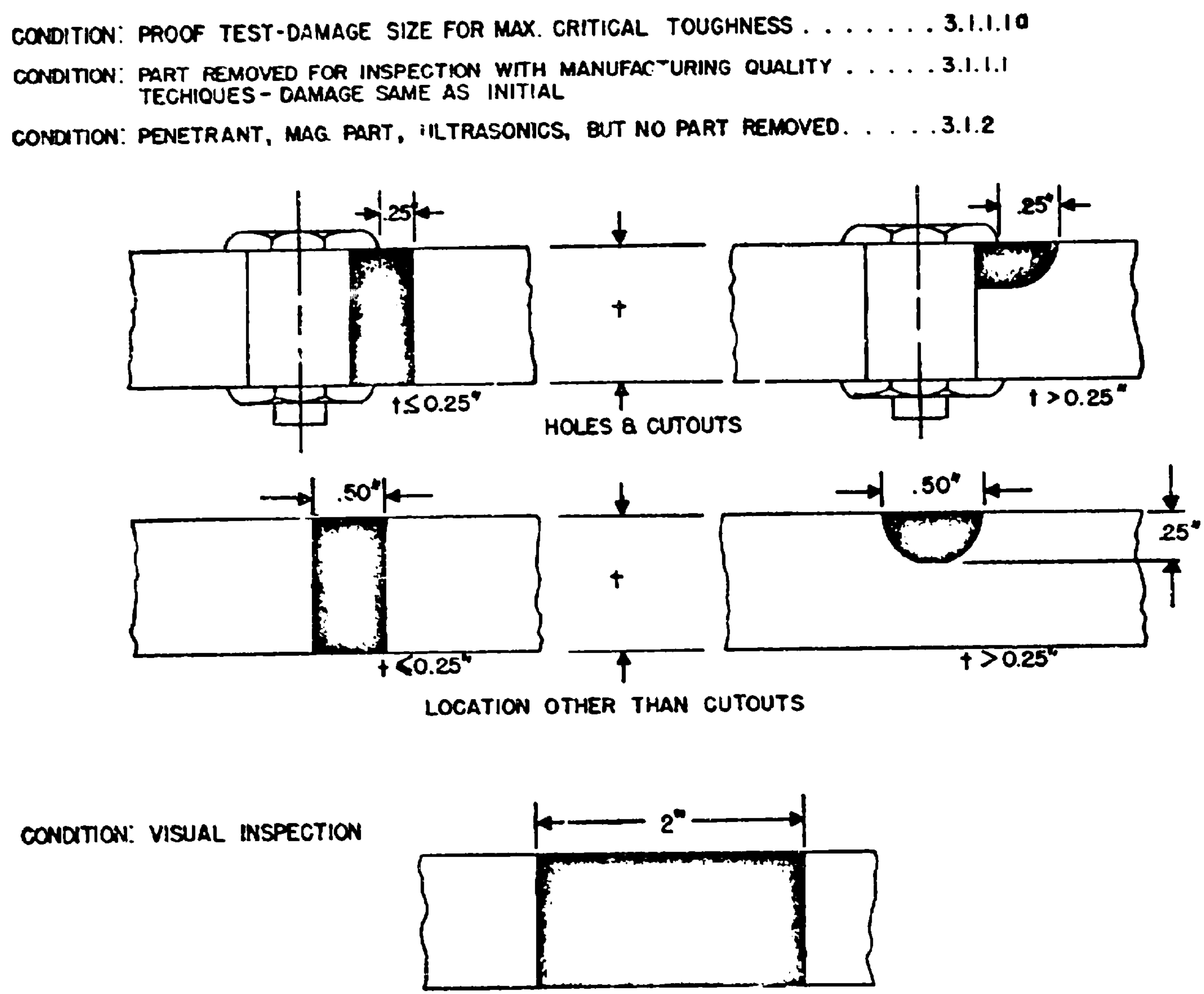


Figure 2.15 Summary of Initial-Flaw Sizes for Structure Qualified as In-Service-Inspectable

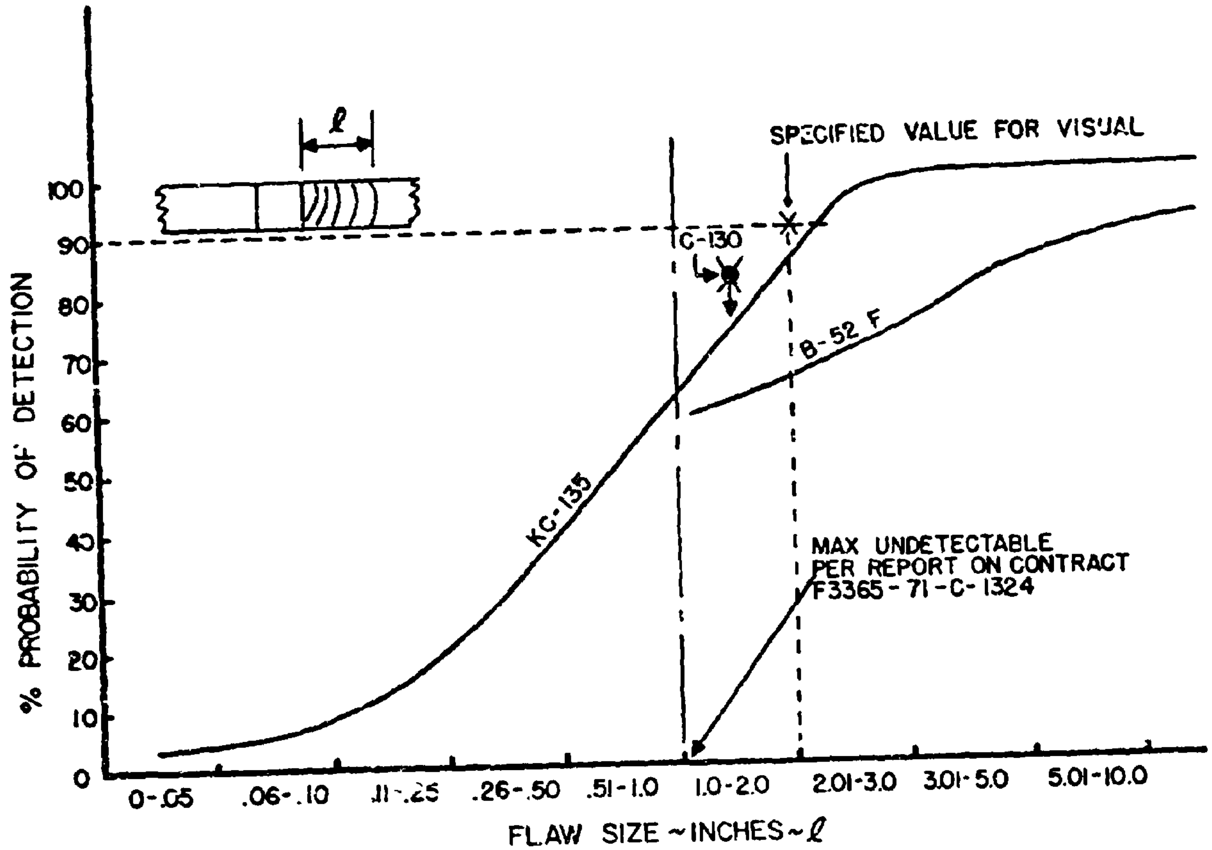


Figure 2.16 Development of Minimum NDI Detection for Visual Inspection

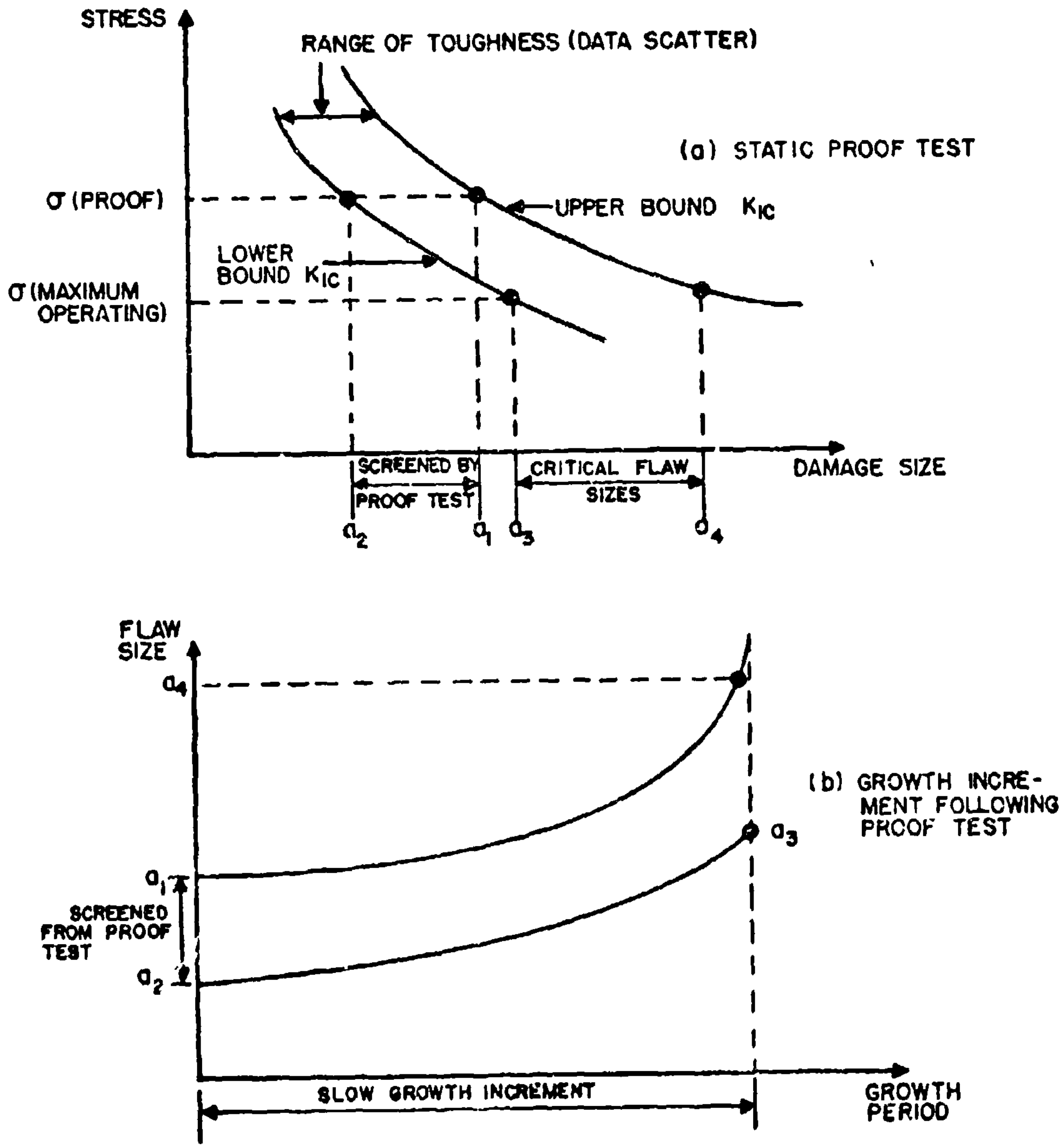


Figure 2.17 Schematic of Proof Test Concept

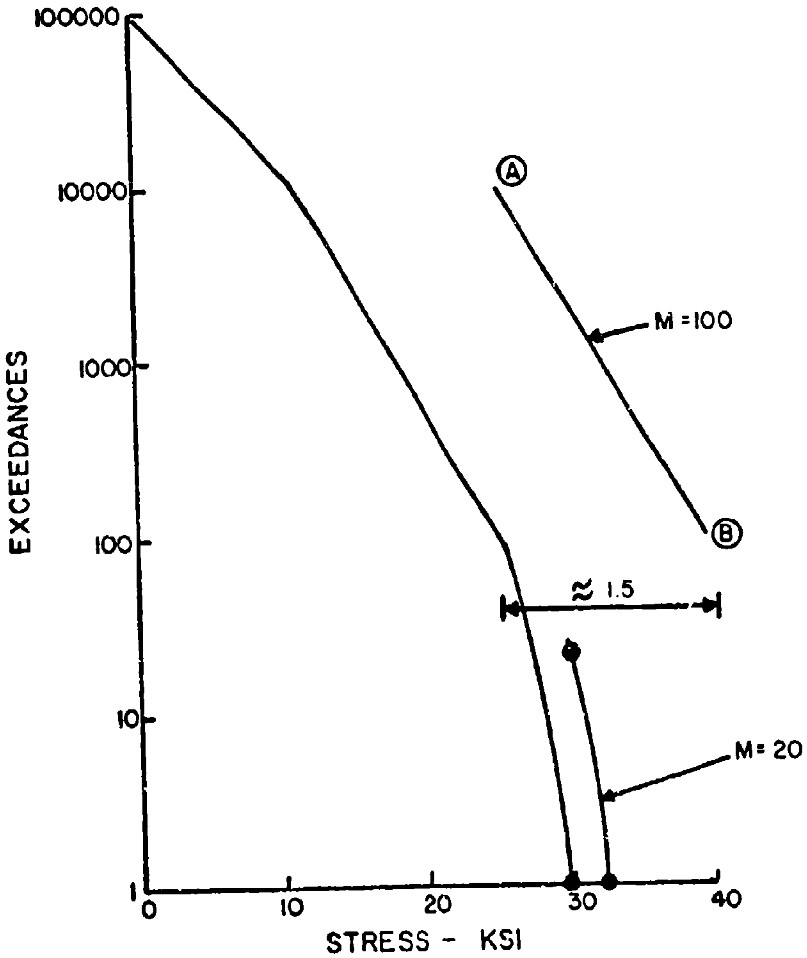


Figure 2.18 Illustration of Procedure to Derive m Factor to Apply to Exceedance Curve

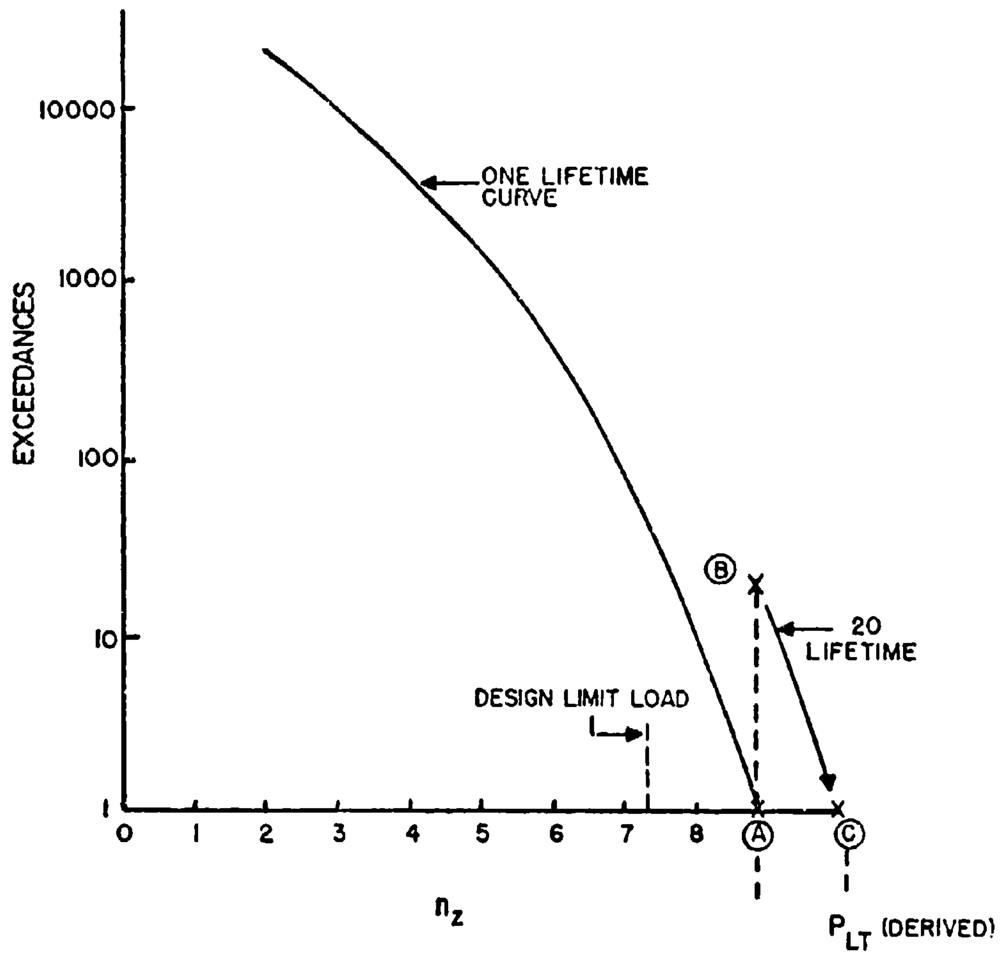


Figure 2.19 Example of the Derivation of P_{XX} from Exceedance Curve

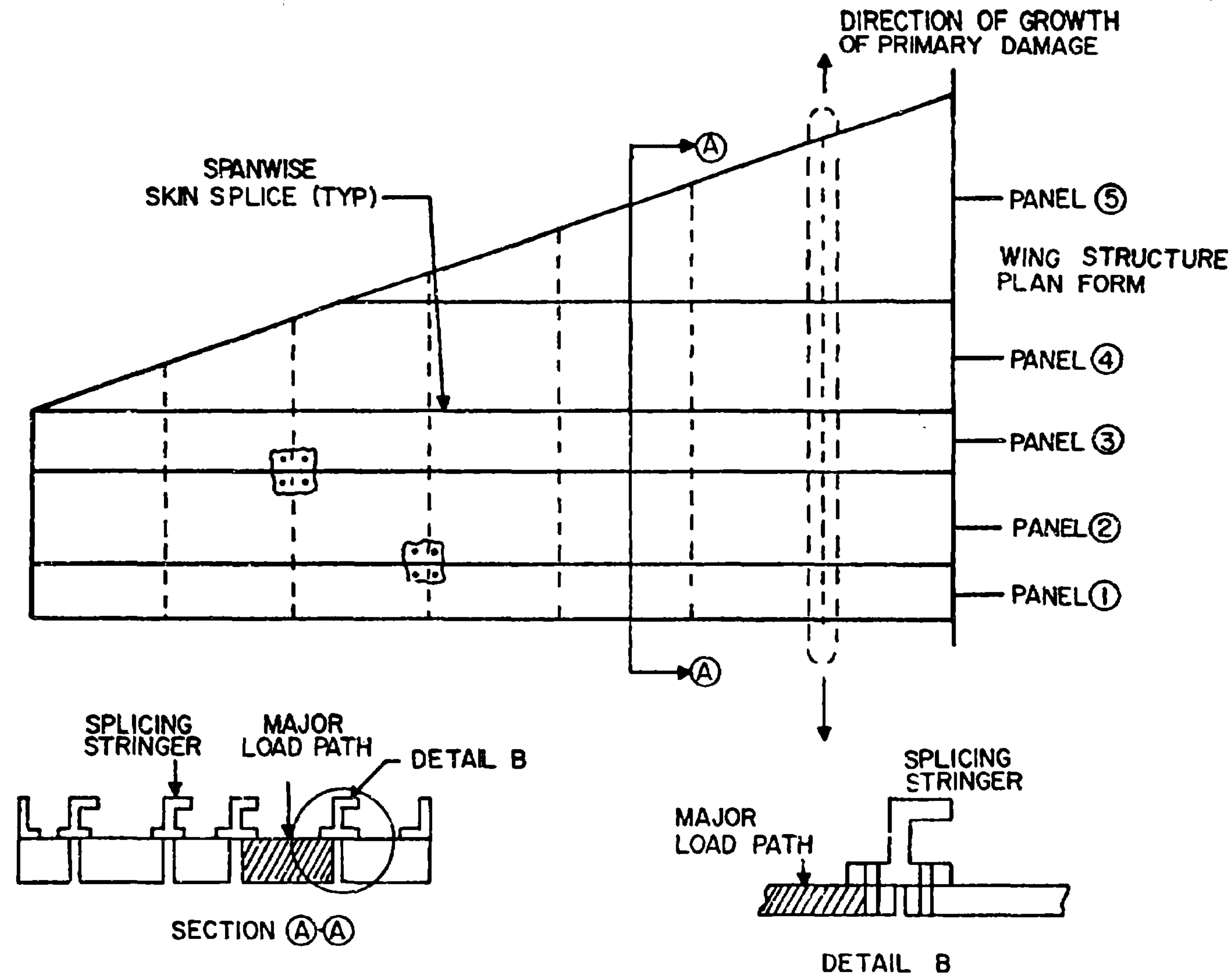


Figure 2.20 Structural Example

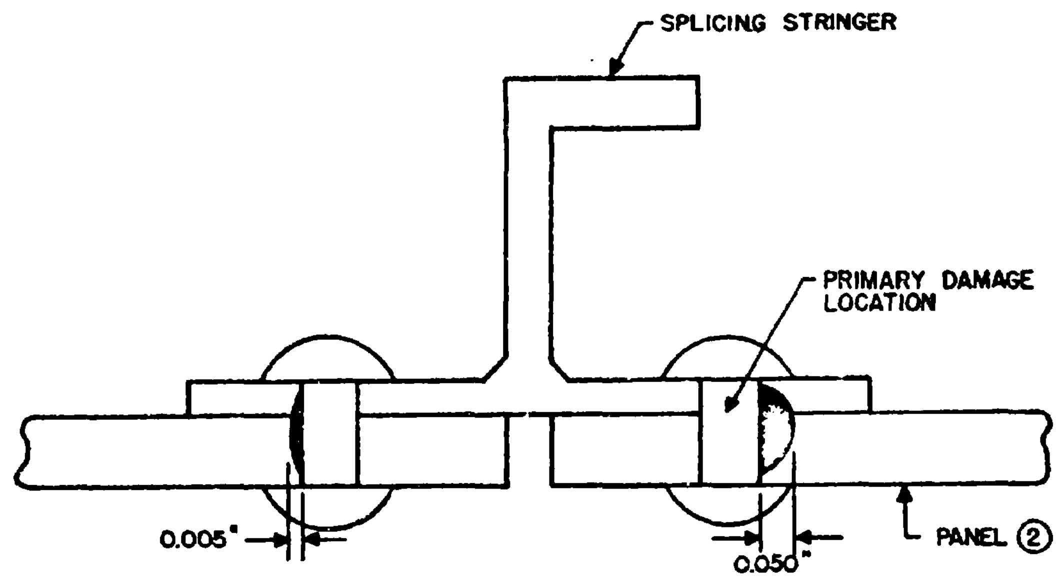


Figure 2.21 Illustration of Initial Flaws for Structure Qualified as Fail-Safe Multiple Load Path

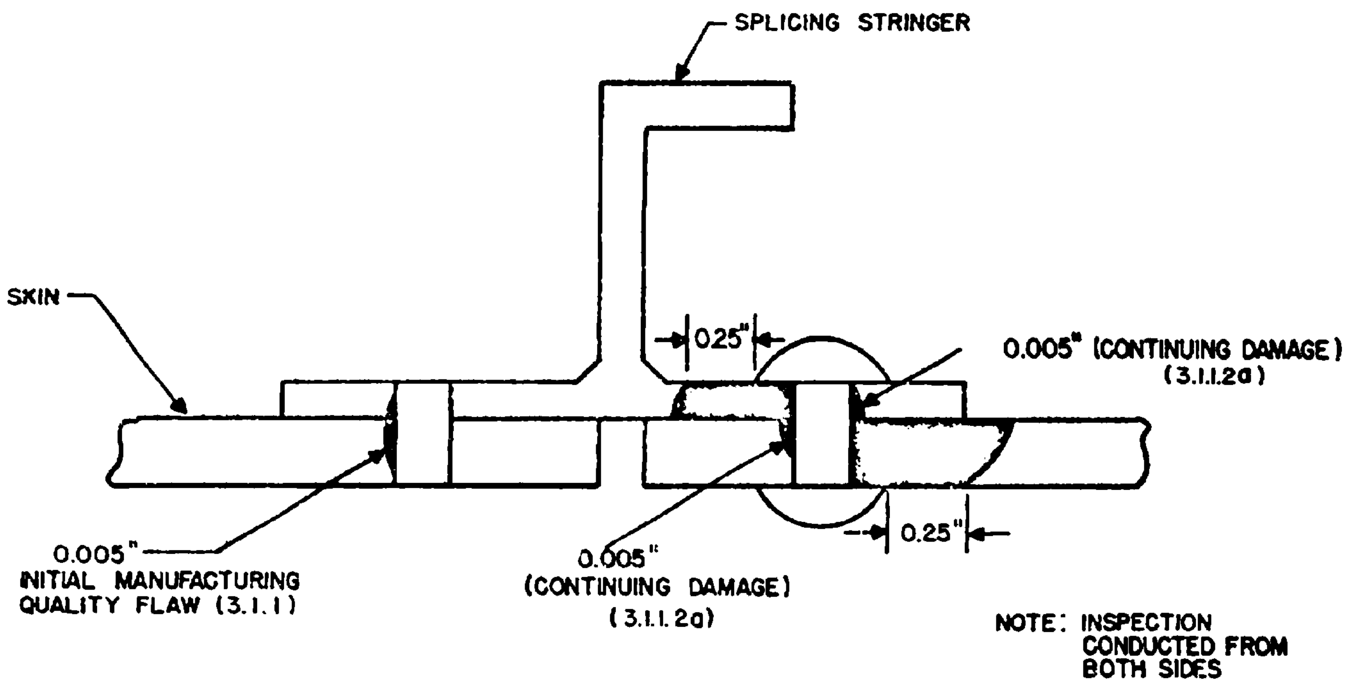


Figure 2.22 Illustration of Primary Damage Following a Depot-Level Penetrant or Ultrasonic Inspection

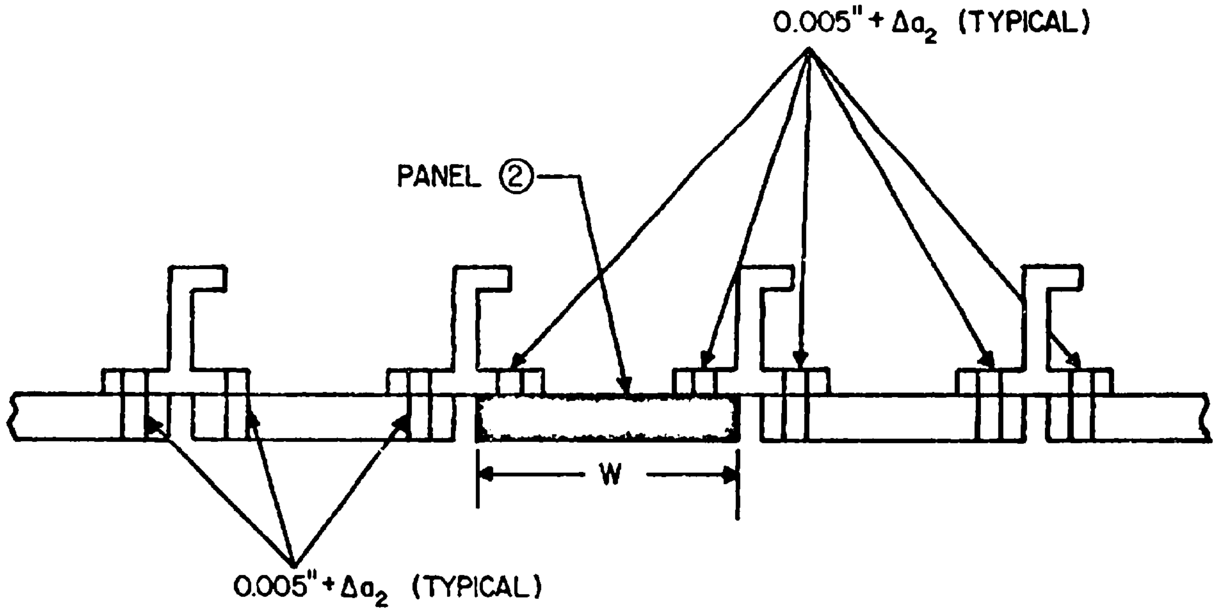


Figure 2.23 Illustration of Primary Damage Assumptions Following the Failure of Major Load Path (Panel 2)

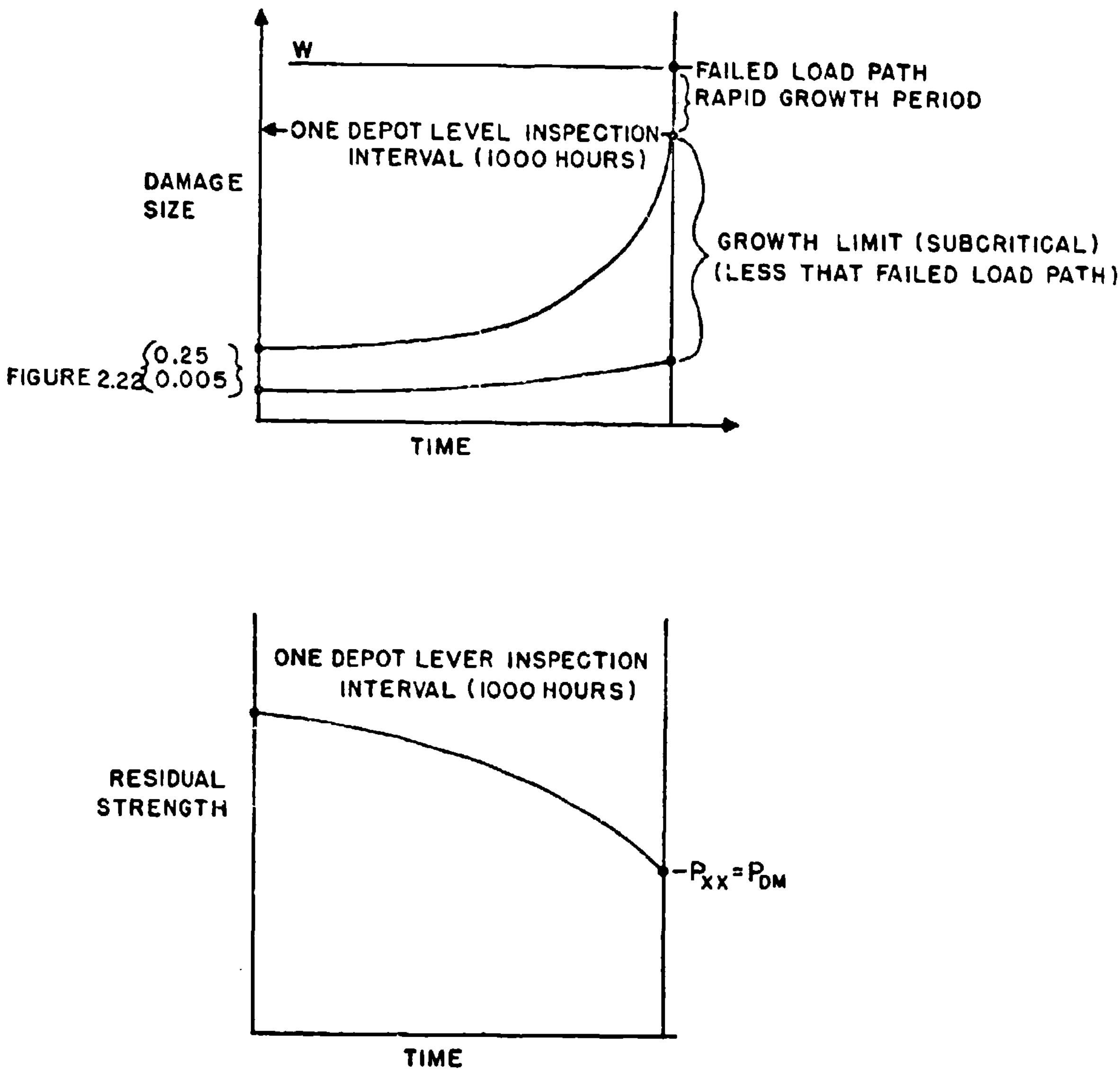


Figure 2.24 Illustration of Residual-Strength and Damage-Growth Limits; Intact Structure Following Depot or Base-Level Inspection for Less-Than-Failed Load Path

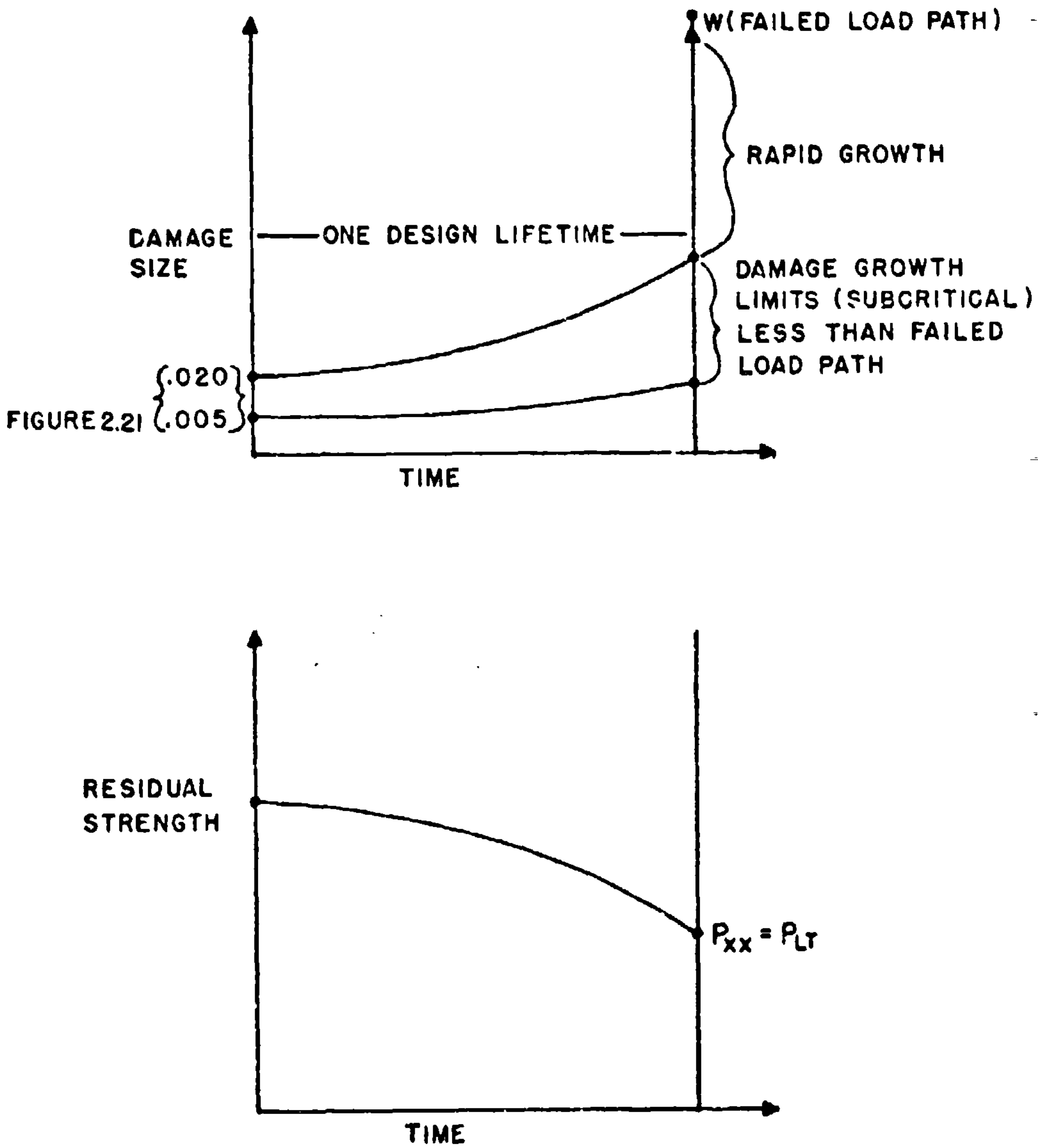


Figure 2.25 Illustration of Residual-Strength and Damage-Growth Limits; Intact Structure for When Depot Inspection Cannot Detect Less-Than-Failed Load Path

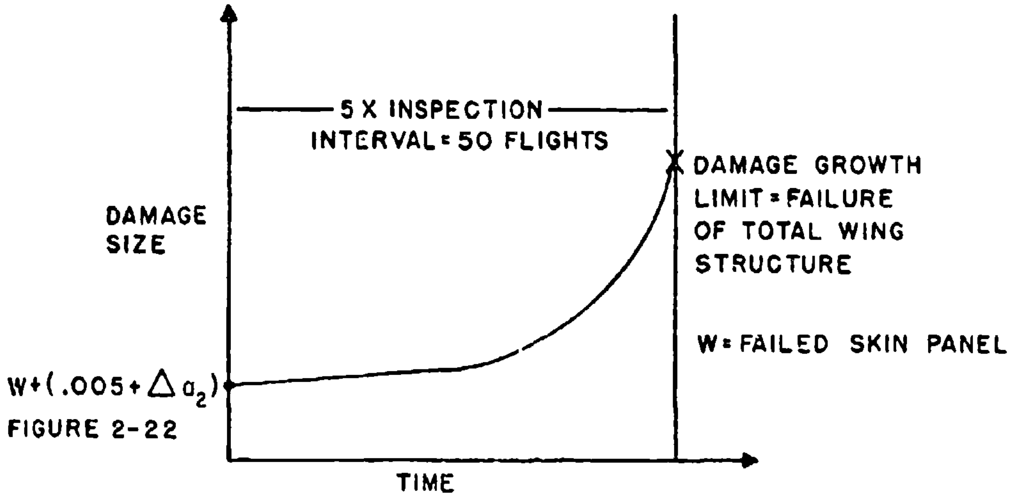


Figure 2.26 Illustration of Damage Growth for Remaining Structure Subsequent to Load-Path Failure

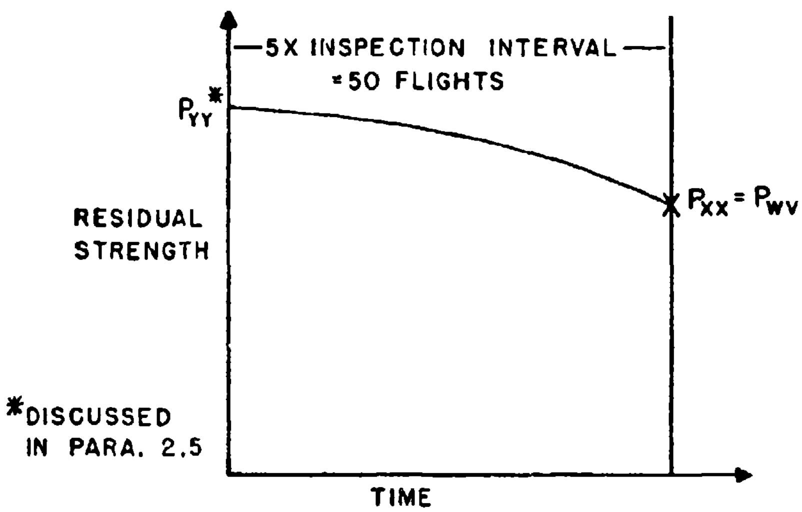


Figure 2.27 Illustration of Residual Strength for Remaining Structure Subsequent to Load-Path Failure

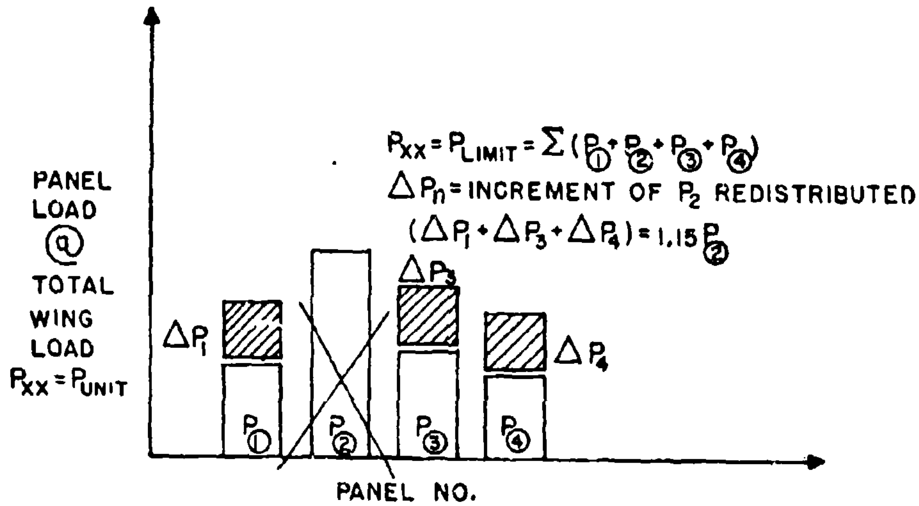


Figure 2.28 Illustration of Redistributed Panel Load P_2 to Adjacent Structure

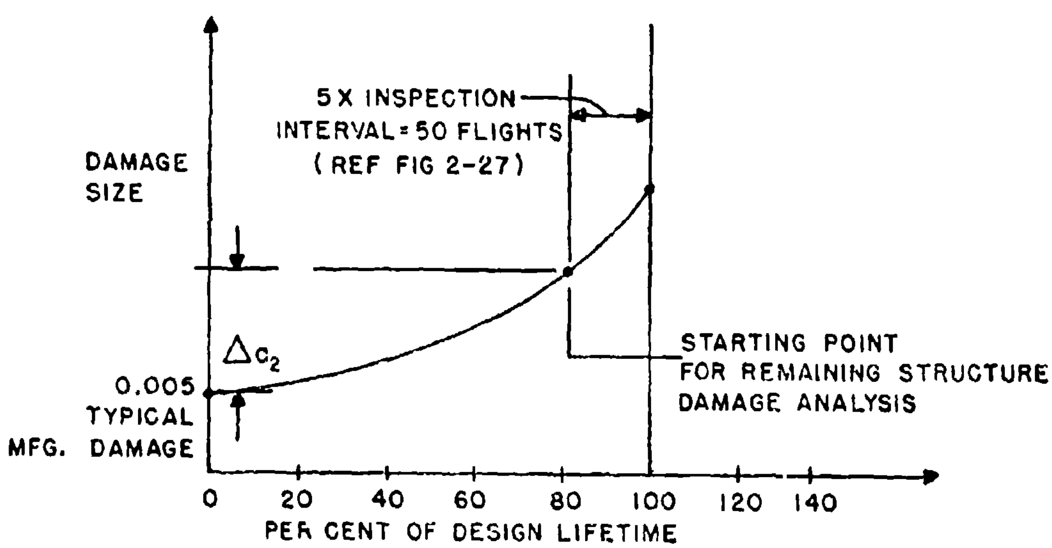


Figure 2.29 Development of Increment of Growth Δa_2 Used in the Analysis of Damage Growth - Remaining Structure Damage - Walk-Around-Visual Inspectable

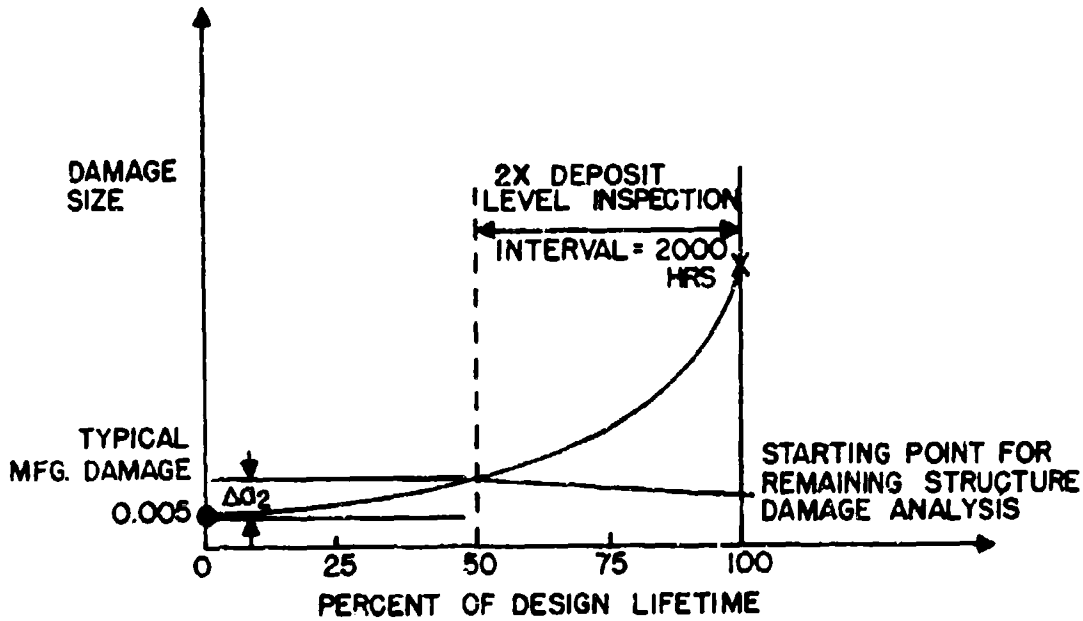


Figure 2.30 Development of Increment of Growth Δa_2 Used in Analysis of Damage Growth - Remaining Structure Damage - Depot-Level-Inspectable

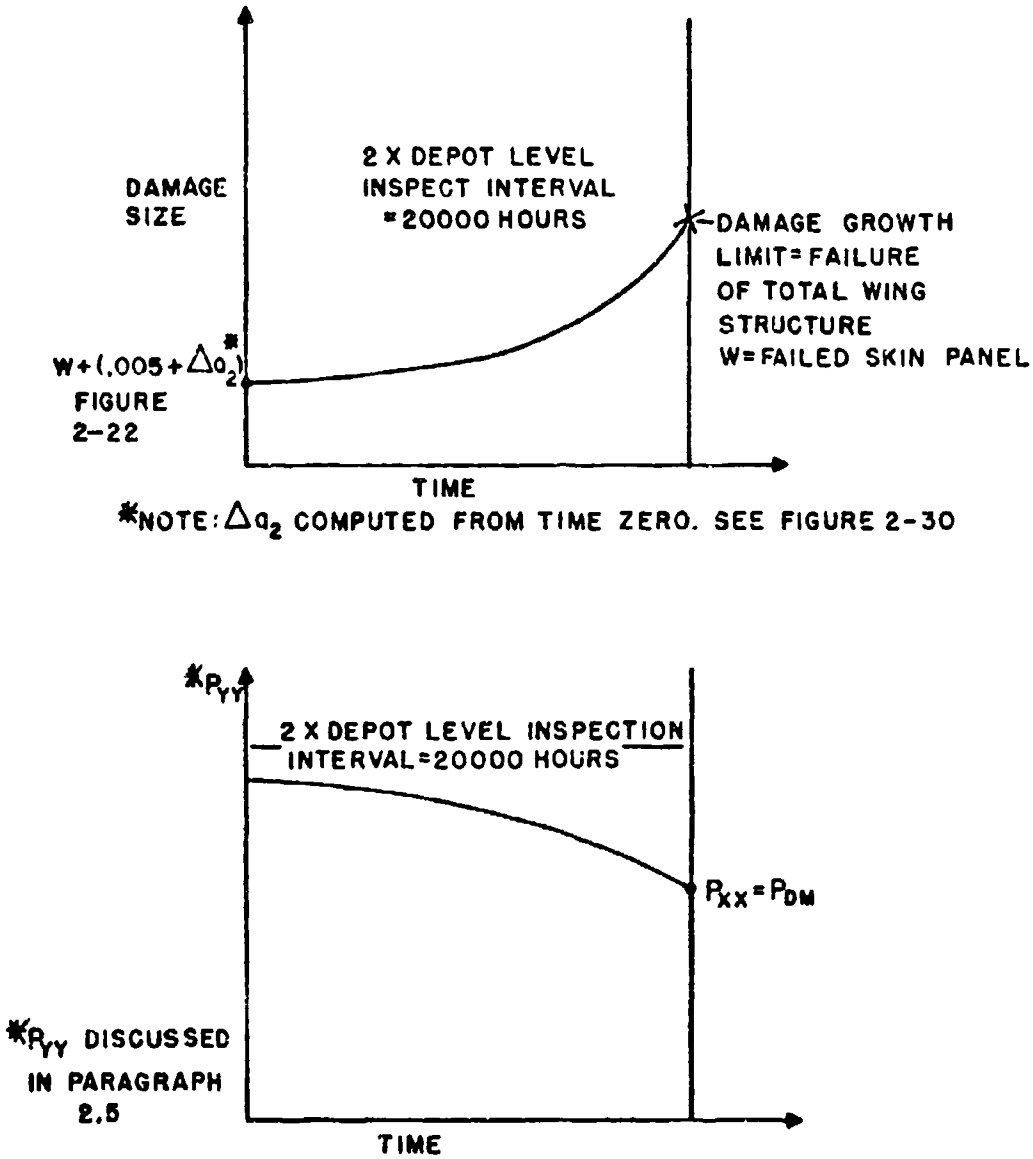


Figure 2.31 Illustration of Damage Growth and Residual-Strength Requirements - Remaining Structure - Depot-Level Inspectable

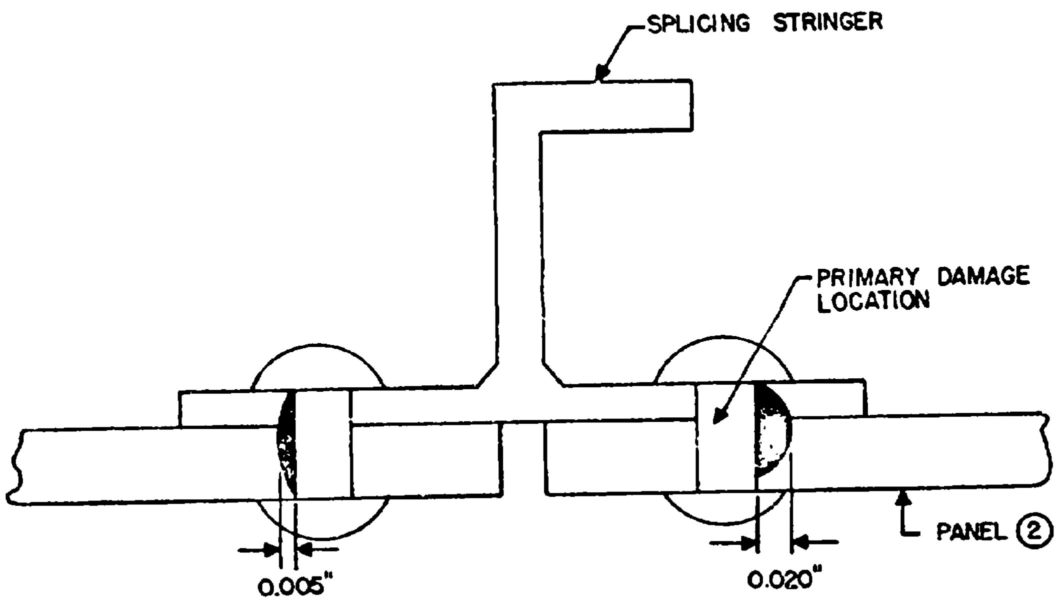
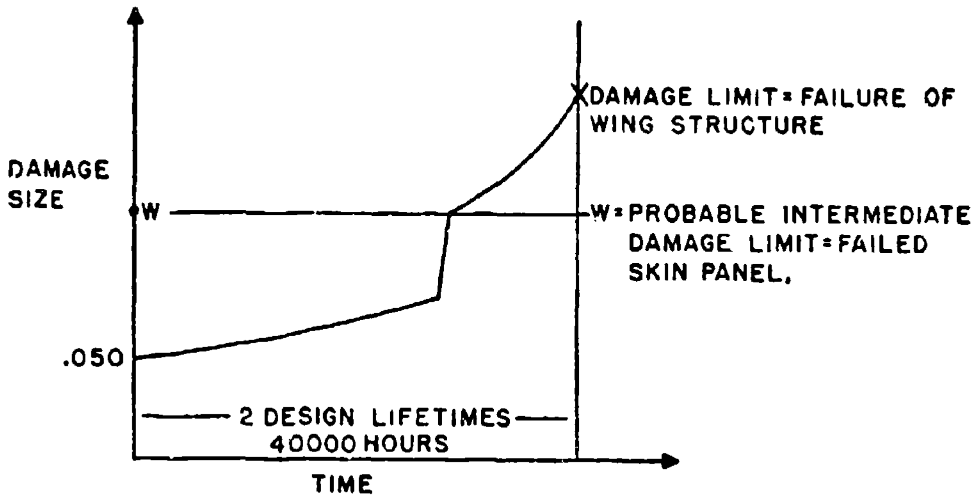


Figure 2.32 Initial-Flaw Assumptions for Example Case Qualified as Slow-Crack-Growth



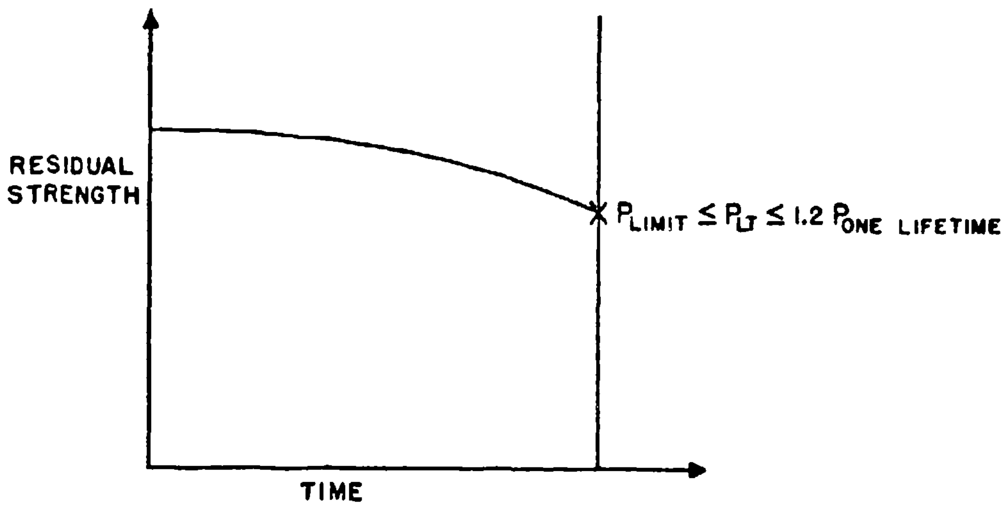


Figure 2.33 Illustration of Damage-Growth and Residual-Strength Requirements for Example Problem Qualified as Slow Crack Growth Noninspectable

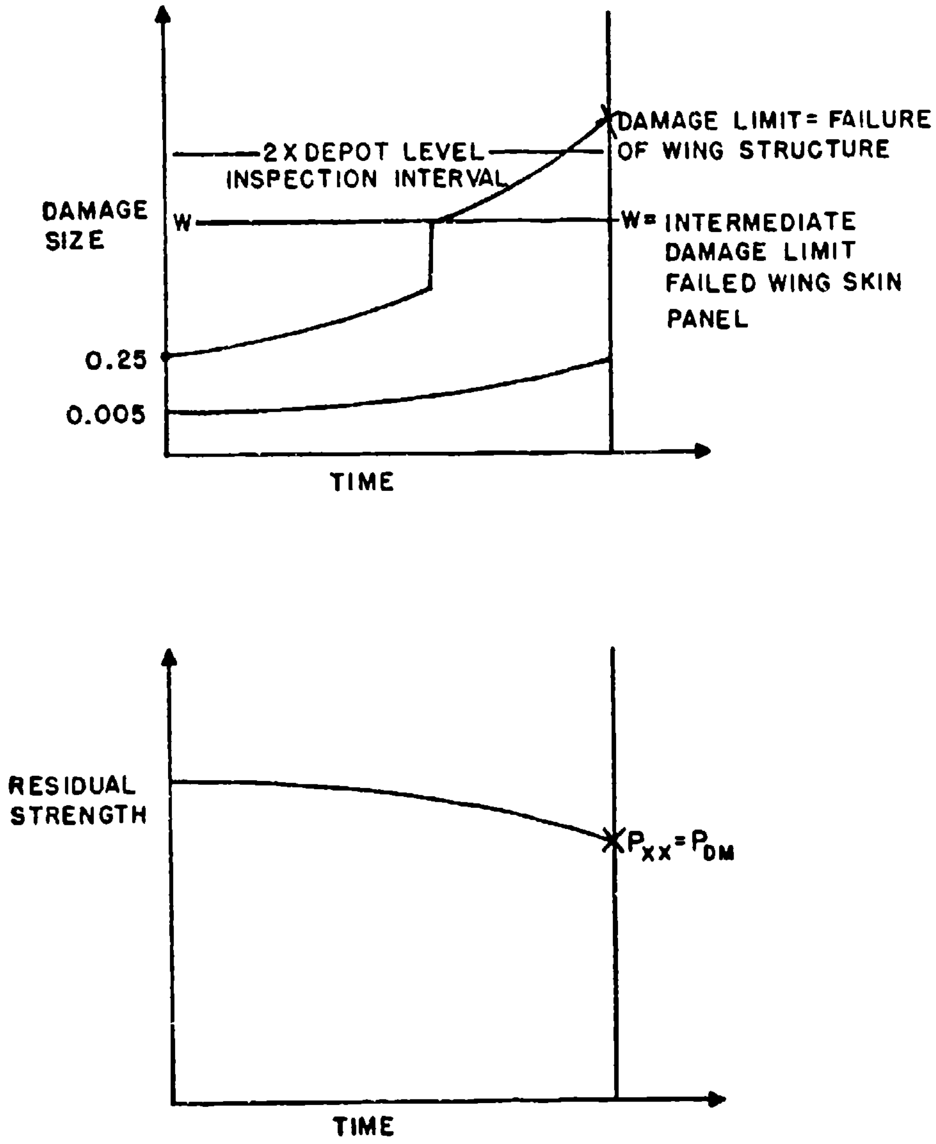


Figure 2.34 Illustration of Damage-Growth and Residual-Strength Requirements for Example Problem Qualified as Depot-Level-Inspectable

CHAPTER 3

Damage Size Considerations
(to be added later)

CHAPTER 4.

Determination of Residual Strength

4.0 DETERMINATION OF RESIDUAL STRENGTH

SECTION PAGE

DETERMINATION OF RESIDUAL STRENGTH
(Cont'd)

LIST OF FIGURES

FIGURE

LIST OF FIGURES
(Cont'd)

LIST OF FIGURES
(Cont'd)

FIGURE

4.1 ELEMENTS OF RESIDUAL STRENGTH ANALYSIS

Residual strength analysis is conducted to determine the capability of a structure containing significant damage to withstand a single, monotonically increasing load for a short time without catastrophic failure. To perform the residual strength analysis, the engineer requires the following elements:

(1) Definition of the required residual strength load, P_{XX}

(2) A failure criterion and associated material properties

(3) Capability to account for geometry of the structure.

4.1.1 Definitions

a. Residual Strength - The strength of a structure can be largely affected by the presence of a crack and is usually substantially lower than the initial strength of the undamaged structure. Therefore, the load carrying capacity of a cracked structure is called the "residual strength" of that structure. The residual strength is a function of

material toughness

crack size and geometr,

structural geometry

When the residual strength of the structure falls below the maximum stress in the service load history, fracture occurs.

b. Crack Growth Instability - A crack in a structure constitutes a high stress concentration. When the load on the panel exceeds a certain limit, the crack will extend. The possible cases of subsequent fracture are shown in Figure 4.1. Under certain conditions, this crack extension immediately will be unstable and the crack will propagate in a fast uncontrollable manner causing total fracture of the component (Figure 4.1a).

Under certain other conditions, crack growth will first be stable with a limited amount of crack growth before arrest (Figure 4.1b). Further, load increase makes it grow more, until crack extension becomes unstable at a higher load.

In the general case, unstable crack propagation results in fracture of the component. Hence, unstable crack growth is what determines the residual strength. Sometimes, however, an unstable crack can be arrested within the component. A further increase of the load is then required to make it unstable again (Figure 4.1c).

c. Stress Intensity Factor, K - Cracks impair the load carrying characteristics of a structure. A crack can be characterized for length and configuration using a structural parameter initially developed (independently) by Irwin and Williams in 1957. The crack parameter, termed the stress intensity factor, K, interrelates the local stresses in the region of the crack with (a) crack geometry (b) structural geometry, and (c) level of load on the structure.

d. Fracture Toughness - The residual strength is reached when the load on a cracked structure reaches a critical value, which depends upon the material and geometry. Since the stress-intensity factor describes the crack-tip stresses, it attains its critical value at the moment of fracture. The critical value of the stress-intensity factor, called the "fracture toughness," is a material property (within certain limits) which can be measured.

4.1.2 Required Residual Strength Load P_{xx}

Safety is assured by designing to specific damage tolerance requirements in which initial damage is never allowed to grow and reduce the residual static strength of the structure below a prescribed level, P_{xx}, throughout the life of the aircraft. P_{xx} is the greater of design limit load or the maximum load that might be encountered during the specified minimum period of unrepaired service usage. If in-service inspections are required to insure safety (e.g. for fail safe designs) then the residual strength level, P_{xx}, is the maximum load likely to occur during the inspection interval. For noninspectable structure, P_{xx} is the maximum load likely to occur during the design lifetime. Transport and bomber type aircraft rarely exceed design limit load during service life. In such cases, the maximum value of P_{xx} would be design limit load. Fighters and attack type aircraft frequently exceed limit load and are designed to sustain P_{xx} in excess of design limit.

4.1.3 Failure Criteria

Failure criteria relate the material properties of the structure to the residual strength. Different criteria come into play depending on whether the material is brittle or ductile. Plasticity and mixed mode loading are also becoming more significant. In order to permit direct transfer from laboratory size specimens to full scale structural behavior, the fracture criteria should be independent of specimen geometry.

4.1.3.1 Critical Stress

Typically, when analyzing built up structure, the residual strength of stiffeners is based upon a criterion which assumes that failure occurs at the ultimate strength of the stringer. Thus, the failure criterion becomes simply

$$\sigma_f = F_{tu}$$

For stringer critical structure this is the dominant failure mode.

4.1.3.2 Critical Stress Intensity Factor (Fracture Toughness)

The stress intensity factor characterizes the entire stress field at the crack tip. It is assumed that fracture occurs when the crack-tip stress intensity factor exceeds some critical value.

The critical K for fracture is denoted as K_{Ic} for plane-strain conditions and K_c for plane-stress conditions. Within the limitations discussed in subsequent sections, K_{Ic} and K_c can be considered as a material property called fracture toughness (with adjectives plane strain or plane stress, respectively). They can be determined by experiment. By equating the value of the stress intensity to K_{Ic} or K_c, the fracture stress (or critical crack size at a given stress) can be calculated for any crack configuration for which an expression for the stress-intensity factor is known.

4.1.3.3 Crack Growth Resistance

The crack growth resistance curve approach has particular application to structures and materials which exhibit a significant amount of slow stable tear. The failure criterion requires that two conditions be met for instability

$$K_S \geq K_R \qquad \frac{\partial K_S}{\partial a} \geq \frac{\partial K_R}{\partial a}$$

where K_R is the resistance to crack extension for the material and K_S is the stress intensity factor for the given structural configuration. These two conditions are met when the K_S and K_R curves become tangent to one another. This is schematically presented in Figure 4.2.

4.1.4 Geometric Considerations

The structural configuration essentially determines the complexity of the residual strength analysis. Typical structural parameters which must be considered are:

a. Type of Construction

1. Monolithic (Unreinforced/Forgings)
2. Skin (Longerons, stringe
3. Integrally Stiffened
4. Planked
5. Layered (Honeycomb/Laminated)

b. Panel Geometry

1. Planform
2. Curvature
3. Stiffener Spacing and Orientation
4. Attachments (Spar Caps, Webs, Frames, etc.)

c. Details of Construction

1. Stiffener Geometry (hat, Z, Channel, etc.)
2. Attachment Details (Bolted, Riveted, Welded, etc.)
3. Fastener Flexibility
4. Eccentricity

Ideally, the residual strength analysis will take all these parameters into consideration. In practice, many are treated empirically and others are not considered except in extremely detailed finite element analyses.

4.1.5 The Residual Strength Diagram

The residual strength diagram is basically a plot of the fracture stress as a function of crack size for the given structural configuration. For single load path structure, the residual strength diagram consists of a single curve as shown in Figure 4.3 for an unstiffened panel.

For stiffened skin construction, which has crack arrest capability, generation of the residual strength diagram is considerably more complex and must be performed in steps.

Consider an axially loaded skin-stringer combination with longitudinal stiffening (Figure 4.4, top). The displacements of adjacent points in skin and stringers will be equal. (If skin and stringers are of the same material, the stresses in the two will also be equal). Let a transverse crack develop in the skin. This will cause larger displacements in the skin. The stringers have to follow this larger displacement. As a result, they take on load from the skin, thus decreasing the skin stress at the expense of higher stringer stress. Consequently, the displacements in the cracked skin will be smaller than in an unstiffened plate with the same size of crack. This implies that the stresses are lower and that the stress-intensity factor is lower. The closer the stringers are to the crack, the more effective is the load transfer.

If the stress intensity for a small central crack in an unstiffened plate is given by $K = \sigma\sqrt{\pi a}$, the stress intensity for the stiffened plate will be $K = \beta\sigma\sqrt{\pi a}$. The reduction factor, $\beta = K/\sigma\sqrt{\pi a}$, will decrease when the crack tip approaches a stringer. Since the stringers take load from the skin, their stress will increase from σ to $L\sigma$, where L increases when the crack tip approaches the stringer. Obviously, $0 < \beta \leq 1$, and $L \geq 1$. These values depend upon stiffening ratios, the stiffness of the attachment, and the ratio of crack size to stringer spacing. As will be shown in the next section, β and L can be readily calculated. For a qualitative discussion, it may suffice that β and L vary as shown diagrammatically in Figure 4.4.

The residual strength diagram of a simple panel with two stringers and a central crack now can be constructed. Recall (Figure 4.1) that a crack in plane stress starts propagating slowly at $K_i = \sigma\sqrt{\pi a_i}$ and becomes unstable at $K_c = \sigma\sqrt{\pi a_c}$. The residual-strength behavior for a sheet without stringers is as shown in Figure 4.5a. There is a line for the onset of crack growth given by $\sigma_i = K_i/\sqrt{\pi a_i}$ and a line for fracture instability given by $\sigma_c = K_c/\sqrt{\pi a}$.

When the panel is stiffened by stringer, the stress intensity is reduced to $K = \beta\sigma\sqrt{\pi a}$, where $\beta < 1$ as shown in Figure 4.4. As a result, both the stress for slow stable crack growth, σ_{is}, and the stress for unstable crack growth, σ_{cs}, for the stiffened panel are given by $\sigma_{is} = K_i/\beta\sqrt{\pi a_i}$ and $\sigma_{cs} = K_c/\beta\sqrt{\pi a_c}$, respectively. Hence, these events take place at higher stresses in the stiffened panel than in the unstiffened panel. T means that the lines in Figure 4.5a, are raised by a factor $1/\beta$ for the case of the stiffened panel, as depicted in Figure 4.5b. Since β decreases if the crack approaches the stringer, the curves in Figure 4.5b turn upward for crack sizes on the order of the stringer spacing.

The possibility of stringer failure should be considered also. The stringer will fail when its stress reaches the ultimate tensile stress (UTS). As the stringer stress is $L\sigma$, where σ is the nominal stress in the panel away from the crack, failure will occur at σ_{sf}, given by $L\sigma_{sf} = \sigma_{uts}$. Using L, as depicted in Figure 4.4, the panel stress at which stringer failure occurs is shown in Figure 4.5c.

THE STRINGER MAY YIELD BEFORE IT FAILS. THIS MEANS THAT ITS CAPABILITY TO TAKE OVER LOAD FROM THE CRACKED SKIN DECREASES. AS A RESULT, β WILL BE HIGHER AND L WILL BE LOWER. THE STRESS-INTENSITY ANALYSIS SHOULD ACCOUNT FOR THIS EFFECT.

Figure 4.6 shows the residual strength diagram of the stiffened panel. It is a composite of the three diagrams of Figure 4.5. In case the crack is still small at the onset of instability ($2a \ll 2s$, where $2s$ is stringer spacing), the stress conditon at the crack tip will hardly be influenced by the stringers and the stress at unstable crack-growth initiation will be the same as that of an unstiffened sheet of the same size. When the unstably growing crack approaches the stiffener, the load concentration in the stiffener will be so high that the stiffener fails without stopping the unstable crack growth (line ABCD in Figure 4.6).

When the panel contains a crack extending almost from one stiffener to the other ($2a \approx 2s$), the stringer will be extremely effective in reducing the peak stress at the crack tips (β small), resulting in a higher value of the stress at crack-growth initiation at point F in Figure 4.6. With increasing load, the crack will grow stably to the stiffener (line EFGH) and due to the inherent increase of stiffener effectiveness, the crack growth will remain stable. (Actually, no unstable crack growth will occur for crack lengths larger than $2a_2$). Fracture of the panel will occur at the stress level indicated by $\overline{\sigma}$ due to the fact that the stiffener has reached its failure stress and the stress reduction in the skin is no longer effective after stringer failure.

For cracks of intermediate size ($2a = 2a_1$), there w ll be unstable crack growth at a stress slightly above the fracture strength of the unstiffened sheet (point M), but this will be stopped under the stiffeners at

N. After crack arrest, the panel load can be further increased at the cost of some additional stable crack growth until H, where the ultimate stringer load is reached, again at the stress level $\bar{\sigma}$.

For the simple panel in Figure 4.6, the actual residual-strength curve is of the shape indicated by the heavy solid line. This curve contains a horizontal part determined by the intersection of lines e and g. For initial cracks smaller than the stiffener spacing, this flat part constitutes a lower bound of the residual strength.

It has been outlined that β and L depend upon stiffening ratio (Figure 4.4). This implies that the residual strength diagram of Figure 4.6 is not unique. It shows the case, wherein stringer failure is the critical event. For other stiffening ratios, skin failure may be the critical event as depicted in Figure 4.7. Due to a low stringer load concentration, the curves e and g do not intersect. A crack of size $2a_1$ will show stable growth at point B and become unstable at point C. Crack arrest occurs at D from where further slow growth can occur if the load is raised. Finally, at point E, the crack will again become unstable, resulting in panel fracture. Apparently, a criterion for crack arrest has to involve the two alternatives of stringer failure and skin failure, depending upon the relative stiffness of sheet and stringer.

The foregoing clearly shows for crack arrest it is not essential that the crack runs into a fastener hole. Crack arrest is basically a result of the reduction of crack-tip stress intensity due to load transmittal to the stringer.

So far, the discussion has been limited to skin critical and stringer critical configurations. Of course, a third criterion exists which concerns fastener failure. Load transmittal from the skin to the stringer takes place through the fasteners. If the fastener loads become too high, fastener failure may take place by shear. Fastener failure will reduce the effectivity of the stringer and therefore the residual strength will drop. The highest loads will be on the fasteners adjacent to the crack path. The load to fail the fasteners by shear can be calculated and the nominal stress in the panel then gives a third line, h, in the residual strength diagram depicted in Figure 4.8.

At zero crack length the fasteners do not carry any load, so line h tends to infinity for $2a \rightarrow 0$. For the particular case depicted in Figure 4.8, the residual strength is no longer determined by stringer failure solely (dashed horizontal line through point H) but possibly by fastener failure (point K). A crack of length $2a_1$ will show slow growth from E to F and instability from F to G. After crack arrest at G, further slow growth occurs until at K the fasteners fail. The latter will probably cause panel failure, but this cannot be directly determined from the diagram. In fact, a new residual strength diagram has now to be calculated with omission of the first row of rivets at either side of the crack. Fastener failure will affect load transmittal from the skin to the stringer: line f will be lowered, line g will be raised. The intersection H' of the new lines g' and f' may still be above K and hence, the residual strength will still be determined by stringer failure at H'.

In reality, the behavior will be more complicated due to plastic deformation. Shear deformation of the fasteners, hole deformation, and plastic deformation of the stringers will occur before fracture takes place. Plastic deformation always leads to a reduction of the effectivity of the stringer to take load from the skin. This implies that line g will be raised and line f will be lowered. The intersection of the two lines (failure point) will not be affected a great deal, however (compare points H and H´ in Figure 4.8). For this reason the residual strength of a stiffened panel can still be predicted fairly accurately, even if plasticity effects are ignored. Nevertheless, a proper treatment of the problem requires that plasticity effects are taken into account.

The cases considered pertain to cracks between two stiffeners. In practice, however, cracks will usually start at a fastener hole and then there will be a stringer across the crack which will have a high load concentration factor. The problem can be dealt with in a manner similar to a crack between stringer, using either analytical or finite-element procedures. A schematic residual-strength diagram for this case is presented in Figure 4.9. Apart from the curve g for the edge stiffeners, there will now be an additional failure curve k for the central stiffener. Failure of the panel may be determined by the intersection L of curves f and k where the central stringer fails. If that occurs, lines g and f are no longer valid, since both the skin and the edge stiffeners will have to take the extra load from the failed stringer. This will lower lines g and f, to g´ and f´ and in general point H´ will be lower than point L. The latter will have to be checked in a complete analysis.

Due to the high load concentration, the middle stringer will usually fail fairly soon by fatigue and therefore lines g´ and f´, with the middle stringer failed, will have to be used and the residual strength is determined by point H´. (Note that g´, f´, and H´ will have different positions in th absence of the middle stringer; a cracked stringer will induce higher stresses in both the skin and the edge stiffener).

IT FOLLOWS FROM THE FOREGOING DISCUSSIONS THAT IT IS NECESSARY TO ESTABLISH A COMPLETE RESIDUAL-STRENGTH DIAGRAM. THIS IMPLIES THAT THE STRUCTURE MUST BE ANALYZED FOR VARIOUS CRACK SIZES; OTHERWISE, THE BEHAVIOR OF THE STRUCTURE CANNOT BE PROPERLY CHECKED. If, for example, in Figure 4.6 one would only consider one crack size $2a = 2s$, then only the points N and T would be determined. These points give no information on the residual strength which is determined by H. In the following sections, methods to calculate the residual strength diagram will be considered.

4.2 RESIDUAL STRENGTH PREDICTION TECHNIQUES

In general, the prediction of residual strength is based on the determination of the critical value of the stress intensity factor for a given geometry and loading. This value is then equated to the material fracture toughness for the appropriate thickness and orientation. From this relationship, the decay in critical stress can be defined in terms of crack size.

4.2.1 Linear Elastic Fracture Mechanics

The linear elastic fracture mechanics approach to analysis of fracture critical structure essentially consists of two techniques. The first technique relies upon closed form solutions for stress intensity factors (K) for typical crack geometries. These solutions are compiled in handbooks and various reference works, e.g. References 2, 49, 50. These solutions may then be extended to more complex cases through the principles of super position. The finite element methods for developing stress intensity factors offer the advantage of being able to model complex structural geometries and loading systems which is vital when load transfer is important.

4.2.1.1 Stress Intensity Factor, K

For any crack problem, the elastic stresses in the immediate vicinity of the crack tip can be given as

$$\sigma_{ij} = \frac{K_I}{\sqrt{2\pi r}} \; f_{ij}\,(\theta) + \text{nonsingular terms,} \qquad (4\text{-}1)$$

where r and θ are polar coordinates originating at the crack tip.

In the vicinity of the crack tip, the nonsingular terms are small with respect to the singular terms. Hence the stresses may be written

$$\sigma_x = \frac{K_I}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left[1 - \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] ,$$

$$\sigma_y = \frac{K_I}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left[1 + \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] ,$$

$$\sigma_z = 0 \text{ (plane stress) or } \sigma_z = \nu(\sigma_u + \sigma_y) \text{ (plane strain)} ,$$

$$\tau_{xy} = \frac{K_I}{\sqrt{2\pi r}} \sin\frac{\theta}{2}\cos\frac{\theta}{2}\cos\frac{3\theta}{2} , \qquad (4\text{-}2)$$

and

$$\tau_{xz} = \tau_{yz} = 0 ,$$

where x is the direction of the crack, y is perpendicular to the crack in the plane of the plate, and z is perpendicular to the plate surface.

Instead of the stresses, one can also use the displacements for the determination of K. In general, the displacements of the crack edges (crack-opening displacements) are employed. The displacement equations are

$$\text{Mode I} \;-\; u = \frac{K_I}{G}\left[\frac{r}{2\pi}\right]^{\frac{1}{2}} \cos\frac{\theta}{2}\left\{1 - 2\nu + \sin^2\frac{\theta}{2}\right\} ,$$

$$v = \frac{K_I}{G}\left[\frac{r}{2\pi}\right]^{\frac{1}{2}} \sin\frac{\theta}{2}\left\{2 - 2\nu - \cos^2\frac{\theta}{2}\right\} , \qquad (4\text{-}3)$$

$$\text{Mode II} - u = \frac{K_{II}}{G}\left[\frac{r}{2\pi}\right]^{1/2} \sin\frac{\theta}{2}\left[2 - 2\nu + \cos^2\frac{\theta}{2}\right],$$

$$v = \frac{K_{II}}{G}\left[\frac{r}{2\pi}\right]^{1/2} \cos\frac{\theta}{2}\left[2\nu - 1 + \sin^2\frac{\theta}{2}\right], \tag{4-4}$$

where r and θ are polar coordinates measured from the crack tip. Since the above elastic field equations are only valid in an area near the tip of the crack, the application should be restricted to that area.

Three loading modes can be distinguished as shown in Figure 10. The associated stress-intensity factors are K_I, K_{II}, and K_{III}. Mode I is technically the more important. It will be the subject of the discussions. (Combined mode loading is considered in a later section).

The stress-intensity factor can always be expressed as

$$K_I = \beta\sigma\sqrt{\pi a} \tag{4-5}$$

where σ is the nominal stress remote from the crack and a is the crack size. The factor β is a function of crack geometry and of structural geometry. Comparison of Equations (4-1) and (4-5) shows that β must be dimensionless. The dimension of K is ksi $\sqrt{\text{in.}}$ or equivalent.

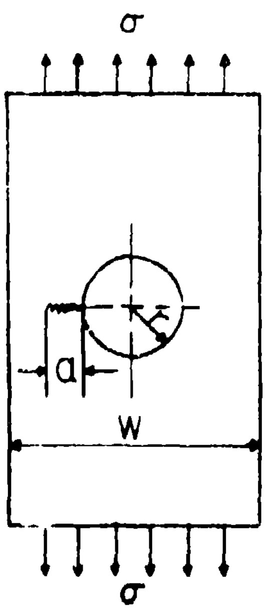
For a central crack of length, 2a, in an infinite sheet, the stress intensity factor may be written

$$K_I = \sigma\sqrt{\pi a} \tag{4-6}$$

comparison with Equation (4-5) shows that for an infinite sheet β is unity. Thus, β may be considered as a correction factor relating the actual stress intensity factor to the central crack in an infinite sheet. The correction factors for various geometrical conditions under a given load condition may be combined in the form of a product to account for the increase or decrease in the stress intensity factor. For example, consider the case of a radial crack in a finite width sheet below. The β factors are collected in Section 4.2.2. Cases under different load conditions may be combined using the principle of super position described in Section 4.2.1.3.

In some cases (e.g., crack-edge loading), Equation (4-5) is not a convenient form for the stress-intensity factor. However, the form of Equation (4-5) is useful in most structural applications; therefore, it will be adopted in these guidelines.

Determination of the stress-intensity factor means calculation of β. For many simple geometries, β will be presented in subsequent sections.



$$K = \beta\sigma\sqrt{\pi a}$$

$$\beta_{FW} = \sqrt{\sec \frac{\pi(a + R)}{W}}$$

$$\beta_H = \frac{0.4367}{0.1623 + \frac{a}{2R}} + 0.6762$$

$$K = \beta_{FW} \cdot \beta_H \cdot \sigma\sqrt{\pi a}$$

Methods to find β for more complex geometries will be presented also.

Since K determines the entire crack-tip stress field, it must be the governing parameter, not only for fracture, but also for other crack-growth processes. The rate of fatigue-crack propagation under cyclic load applications as well as the rate of stress corrosion cracking are a function of K. The higher the stress intensity, the higher is the rate of crack growth. THUS, CRACK GROWTH AND FRACTURE ARE DETERMINED BY THE SAME STRESS FIELD PARAMETER. HENCE, DAMAGE-TOLERANCE CALCULATIONS CAN CONVENIENTLY BE BASED ON THE STRESS-INTENSITY-FACTOR.

4.2.1.2 Closed Form Solutions

As pointed out previously, the establishment of the stress-intensity factor is mainly a determination of β in Equation (4-5). There are several ways to determine the stress-intensity factor.

- Compilations of stress-intensity factors for many different geometries have been made[49,50]. These can be used to find K for relatively simple geometries. (Stress-intensity factors for several structural geometries are also presented in Section 4.3).

- The basic solutions for simple geometries can be derived by means of complex stress functions[1-5]. For finite size bodies, the boundary conditions usually prohibit a closed form solution. In such cases, numerical solutions can be used such as boundary collocation procedures[51].

- Solutions for complicated structural geometries can sometimes be obtained from the basic stress field solutions combined with displacement compatibility requirements for all the structural members involved.

Each of these approaches has its drawbacks. In the first case, the geometries and loadings are restricted to very simple cases. Some of these cases may be extended to other relatively simple cases through super position. This will be discussed in the next section. The complex stress function approach is again very restrictive with regard to geometry and loading. Further, the numerical solution procedures have very little versatility. Consequently, this approach tends to be more academically oriented than useful as an engineering tool. The third approach has been shown by several investigators to be useful in the analysis of built up sheet structure. While these are based on closed form solutions, the actual analyses are computerized for efficient solutions. The essentials of this technique are described below.

Analysis methods for stiffened panels have been developed independently by Romualdi, et al[75], Poe[76,77], Vlieger[73,74], Swift and Wang[78,79], and Creager and Liu[80]. Applications of the stress-intensity-reduction factor, β, and the stringer load concentration factor, L, were proposed by Vlieger[73,74] and Swift and Wang[78,79].

In calculating β and L, two methods can be used, viz, the finite-element method and an analytical method based on closed-form solutions. The analytical method has advantages over the finite-element method in that the effect of different panel parameters on the residual strength of a certain panel configuration can be easily assessed, so that the stiffened panel can be optimized with respect to fail-safe strength. It allows direct determination of the residual-strength diagram. In the case of the finite-element method, a new analysis has to be carried out when the dimensions of certain elements are changed because a new idealization has to be made. An advantage of the finite-element analysis, on the other hand, is that it is relatively easy to incorporate such effects as stringer eccentricity, hole deformation, and stringer yielding. Details of the calculations can be found in the referenced papers.

The basic procedure for the analytical calculation is outlined in Figure 4.11. The stiffened panel is split up into its composite parts, the skin and the stringer. Load transmission from the skin to the stringer takes place through the fasteners. As a result, the skin will exert forces F_1, F_2, etc., on the stringer, and the stringer will exert reaction forces F_1, F_2, etc. on the skin. This is depicted in the upper line of Figure 4.11.

The problem is now reduced to that of unstiffened plate loaded by a uniaxial stress, σ, and fastener forces $F_1 \ldots F_n$. This case can be considered as superposition on three others, shown in the second line of Figure 4.11. Namely:

a. A uniformly loaded cracked sheet.

b. A sheet without a crack, loaded with forces $F_1 \ldots F_n$.

c. A cracked sheet with forces on the crack edges given by the function p(n). The forces p(n) represent the load distribution given by Love[81]. When the slit CD is cut, these forces have to be exerted on the edges of the slit to provide the necessary crack-free edges.

The three cases have to be analyzed. For case a, the stress-intensity factor is $K = \sigma\sqrt{\pi a}$. For case b, K = 0. The stress intensity for case c is a complicated expression that has to be solved numerically.

Compatibility requires equal displacements in sheet and stringer at the corresponding fastener locations. These compatibility requirements deliver a set of n (n = number of fasteners) independent algebraic equations from which the fasteners can be derived.

The number of fasteners to be included in the calculation depends somewhat upon geometry and crack size. According to Swift[82] and shown in Figure 4.12, 15 fasteners at either side of the crack seems to be sufficient to get a consistent result. Similar results were obtained by Sanga[83].

Swift's analysis provides a detailed description of how to incorporate nonelastic behavior in this kind of analysis. The method can account for (1) stiffener flexibility and stiffener bending, (2) fastener flexibility, and (3) biaxiality. Stringer yielding, fastener flexibility, and hole flexibility are lumped together in an empirical equation for fastener deflection.

The effect of fastener flexibility and stiffener bending on β and L is shown in Figure 4.13. Although the effects are quite large, the vertical position of the crossover of stress-intensity curve and stringer stress curve is not affected too much (compare points A and B in Figure 4.13). The level of the crossover determines the residual strength, as pointed out in the previous section. This explains why the residual strength can be reasonably well predicted if flexibility of fasteners is neglected. HOWEVER, FOR APPLICATION OF THE DAMAGE TOLERANCE REQUIREMENTS THERE IS A NEED FOR AN ACCURATE RESIDUAL STRENGTH DIAGRAM. THEREFORE, FASTENER FLEXIBILITY, STRINGER BENDING, AND STRINGER YIELDING WILL HAVE TO BE TAKEN INTO ACCOUNT.

4.2.1.3 Superposition

If the configuration under consideration is not directly presented in stress-intensity factor handbooks[49,50], the stress intensity often can be arrived at by means of a superposition of known solutions[11]. SINCE THE STRESS FIELD EQUATIONS ARE THE SAME FOR ALL MODE I CASES, THE STRESS INTENSITY FOR A COMBINATION OF MODE I LOAD SYSTEMS (p, q, AND r) CAN BE OBTAINED FROM SIMPLE SUPERPOSITION

$$K_I = K_{Ip} + K_{Iq} + K_{Ir} \quad . \tag{4-7}$$

The usefulness of the superposition principle can best be illustrated by means of an example. Figure 4.14 shows a plate without a crack under uniaxial tension. For this case, $K_I = 0$ because there is no singular· . A cut of length 2a is made in the center of the plate. This is allowed if the stresses previously transmitted by the cut material are applied as external stresses to the edges of the slit. This leads to case b in Figure 4.14 where K_{Ib} is still zero, because case b is exactly the same as case a. Case b can be considered a superposition of cases d and e; i.e., a plate with a central crack under tension and a plate with central crack loaded only along the crack edges. Hence,

$$K_{Id} + K_{Ie} = K_{Ib} = 0 \text{ or } K_{Ie} = - K_{Id}. \qquad (4\text{-}8)$$

Since K_{Id} is known to be $K_{Id} = \sigma\sqrt{\pi a}$, it follows that $K_{Ie} = -\sigma\sqrt{\pi a}$. If the direction of the stress in case e is reversed (i.e., crack under internal pressure), the stress intensity is $K_I = \sigma\sqrt{\pi a}$.

Now consider the configuration of Figure 4.15a. This system can be obtained from a superposition of the three other cases shown. From the super-position, it follows that

$$K_{Ia} = K_{Ib} + K_{Id} - K_{Ie} . \qquad (4\text{-}9)$$

Since it is obvious that $K_{Ia} = K_{Ie}$, the stress intensity is

$$K_{Ia} = \tfrac{1}{2}\left(K_{Ib} + K_{Id}\right) = \tfrac{1}{2}\left(1 + \frac{W}{2a}\right) \sigma \sqrt{\pi a} . \qquad (4\text{-}10)$$

A more complex example is illustrated in Figure 4.16. This figure shows a two step approach for obtaining an approximate solution for intermediate values of load transfer at a pen loaded hole. The first step is to obtain the stress intensity factor for the case in which the pin reacts the entire load. This is obtained by noting that superposition of $K_B + K_D$ yields twice the desired solution K_A since K_E is merely the reverse of K_A. Having obtained an expression for K_A, it then becomes a simple matter to obtain K_F by superposition of K_A and K_B with the appropriate choice of σ in each.

4.2.1.4 Finite Element Methods

In all cases where an expression for the stress-intensity factor cannot be obtained from existing solutions, finite-element analysis can be used to determine K[52-56]. Certain aircraft structural configurations have to be analyzed by finite-element techniques because of the influence of complex geometrical boundary conditions or complex load transfer situations. In the case of load transfer, the magnitude and distribution of loadings may be unknown. With the application of finite-element methods, the required boundary conditions and applied loadings must be imposed on the model.

Complex structural configurations and multicomponent structures present special problems for finite-element modeling. These problems are associated with the structural complexity. When they can be solved, the stress-intensity factor is determined in the same way as in the case of a simpler geometry. This section deals with the principles and procedures that permit the determination of the stress-intensity factor from a finite-element solution. Each procedure will be discussed. Then each will all be applied to derive a K solution for the case of a through crack at a hole. This will allow a judgement of the relative accuracy of the various procedures.

Usually quadrilateral, triangular, or rectangular constant-strain elements are used, depending on the particular finite-element structural analysis computer program being used. For problems involving holes or other stress concentrations, a fine-grid network is required to accurately model the hole boundary and properly define the stress and strain gradients around the hole or stress concentration.

WITHIN THE FINITE-ELEMENT GRID SYSTEM OF THE STRUCTURAL PROBLEM, THE CRACK SURFACE AND LENGTH MUST BE SIMULATED. USUALLY, THE LOCATION AND DIRECTION OF CRACK PROPAGATION IS PERPENDICULAR TO THE MAXIMUM PRINCIPAL STRESS DIRECTION. IF THE MAXIMUM PRINCIPAL STRESS DIRECTION IS UNKNOWN, THEN AN UNCRACKED STRESS ANALYSIS OF THE FINITE-ELEMENT MODEL SHOULD BE CONDUCTED TO ESTABLISH THE LOCATION OF THE CRACK AND THE DIRECTION OF PROPAGATION.

The crack surfaces and lengths are often simulated by double-node coupling of elements along the crack line. Progressive crack extension is then simulated by progressively "unzipping" the coupled nodes along the crack line. Because standard finite-element formulations do not treat singular stress behavior in the vicinity of the ends of cracks, special procedures must be utilized to determine the stress-intensity factor. Three basic approaches to obtain stress-intensity factors from finite-element solutions have been rather extensively studied in the literature. These approaches are as follows:

a. Direct Method. The numerical results of stress, displacement, ment, or crack-opening displacement are fitted to analytical forms of crack-tip-stress-displacement fields to obtain stress-intensity factors.

b. Indirect Method. The stress intensity follows from its relation to other quantities such as compliance, elastic energy, or work energy for crack closure.

c. Cracked Element. A hybrid-cracked element allowing a stress singularity is incorporated in the finite-element grid system and stress-intensity factors are determined from nodal point displacements along the periphery of the cracked element.

Each of the above approaches can be applied to determine both Mode I and Mode II stress-intensity factors. Application of the methods has been limited to two-dimensional planar problems. The state of the art for treating three-dimensional structural crack problems is still a research area.

4.2.1.4.1 Direct Methods

The direct methods use the results of the general elastic solutions to the crack-tip stress and displacement fields. For the Mode I, the stresses can always be described by Equation 4.1.

If the stresses around the crack tip are calculated by means of finite-element analysis, the stress-intensity factor can be determined as

$$K_I = \sigma_{ij} \frac{\sqrt{2\pi r}}{f_{ij}(\theta)} \ . \qquad (4\text{-}11)$$

By taking the stress calculated for an element not too far from the crack tip, the stress intensity follows from a subsitution of this stress and the r and θ of the element into Equation (4-11). This can be done for any element in the crack tip vicinity.

Ideally, the same value of K_I should result from each substitution; however, Equations (4-2) are only valid in an area very close to the crack tip. Also at some distance from the crack tip, the nonsingular terms [Equation (4-1)] should be taken into account. Consequently, the calculated K differs from the actual K. The result can be improved[52] by refining the finite-element mesh or by plotting the calculated K as a function of the distance of the element to the crack tip. The resulting line should be extrapolated to the crack tip, since Equations (4-2) are exact for $r = 0$. USUALLY, THE ELEMENT AT THE CRACK TIP SHOULD BE DISCARDED. SINCE IT IS TOO CLOSE TO THE SINGULARITY, THE CALCULATED STRESSES ARE LARGELY IN ERROR. AS A RESULT, EQUATION (4-11) YIELDS A K VALUE THAT IS MORE IN ERROR THAN THOSE FOR MORE REMOTE ELEMENTS, DESPITE THE NEGLECT OF THE NONSINGULAR TERMS.

Instead of the stresses, one can also use the displacements for the determination of K. In general, the displacements of the crack edges (crack-opening displacements) are employed. The displacement equations are given by Equations (4-3) where r and θ are polar coordinates of nodal-point displacements measured from the crack tip. Since the above elastic field equations are only valid in an area near the tip of the crack, the application should be restricted to that area.

The usual approach is to calculate the stress-intensity factors, K_I and K_{II}, by the relative opening and edge-sliding displacements of nodes along the crack surface which has been simulated by coupled-nodal point unzipping.

4.2.1.4.2 Compliance Method

The compliance method makes use of the relation between K and the compliance, C. The compliance is defined as the inverse of the stiffness of the system; i.e., C = v/P, where P is the applied load and v is the displacement of the load application points. The stress-intensity factor is a function of the derivative of the compliance with crack size[11],

$$K = P\sqrt{\frac{E}{2B}\frac{\partial C}{\partial a}} , \qquad (4\text{-}12)$$

where B is the plate thickness and E is Young's modulus.

The compliance is calculated from the finite-element analysis for a range of crack sizes. Differentiation with respect to crack size gives K through Equation (4-12). This can be achieved by solving the same problem for a number of crack sizes (which is facilitated by a computer program with a self-generating mesh system), or by successively unzipping nodes in the cracked section.

The advantage of the method is that a fine mesh is not necessary, since accuracy of crack-tip stresses is not required. A disadvantage is that differentiation procedures always introduce errors.

4.2.1.4.3 Work-Energy Method

The work-energy method determines K from the crack-tip closing work. The work done by the closing forces at the crack tip can be shown(11) to be equal to the energy-release rate, G. The crack-tip closing work can be calculated by uncoupling the next nodal point in front of the crack tip and by calculating the work done by the nodal forces to close the crack to its original size. The stress-intensity factor is found from the relations

$$K_I^2 = EG_I \qquad \text{(plane stress)}$$

$$K_I^2 = EG_I(1-\nu^2) \quad \text{(plane strain)} \qquad (4\text{-}13)$$

$$K_{II}^2 = EG_{II}/(1-\nu^2) .$$

The concept is that if a crack were to extend by a small amount, Δa, the energy absorbed in the process is equal to the work required to close the crack to its original length. The general integral equations for strain energy release rates for Modes I and II deformations are

$$G_I = \lim_{\Delta a \to 0} \frac{1}{2\Delta a} \int_0^{\Delta a} \sigma_y \, (\Delta a - r, 0) \; v(r, \pi) \; dr \quad ,$$

$$(4\text{-}14)$$

$$G_{II} = \lim_{\Delta a \to 0} \frac{1}{2\Delta a} \int_0^{\Delta a} \tau_{xy} \, (\Delta a - r, 0) \; u(r, \pi) \; dr \quad .$$

The significance of this approach is that it permits an evaluation of both K_I and K_{II} from the results of a single analysis.

In finite-element analysis, the displacements have a linear variation over the elements and the stiffness matrix is written in terms of forces and displacements at the element corners or nodes. Therefore, to be consistent with finite-element represen ation, the approach for evaluating G_I and G_{II} is based on the nodal-point forces and displacements. An explanation of application of this work-energy method is given with reference to Figure 4.17. The crack and surrounding elements are a small segment from a much larger finite-element model of a structure. In terms of the finite-element representation, the amount of work required to close the crack, Δa, is one-half the product of the forces at nodes c and d which are required to close these nodes. The expressions for strain energy-release rates in terms of nodal-point displacements and forces are (see Figure 4.17 for notations)

$$G_I = \lim_{\Delta a \to 0} \frac{1}{2\Delta a} \bar{F}_c (v_c - v_d) \quad ,$$

$$(4-15)$$

$$G_{II} = \lim_{\Delta a \to 0} \frac{1}{2\Delta a} \bar{T}_c (u_c - u_d) \quad .$$

With reference to Figure 4.17, the forces at nodes c or d are determined in the following way. The normal and shear stresses near the crack tip vary as $1/r^{\frac{1}{2}}$. Thus, the force over a given length, Δa, is

$$F(\Delta a) = A_1 \int_0^{\Delta a} \frac{dr}{\sqrt{a}} = 2A_1 \, (\Delta a)^{\frac{1}{2}} \quad , \tag{4-16}$$

and for the length, ℓ_2, $F(\ell_2) = 2A_1\ell_2^{\frac{1}{2}}$. Therefore,

$$F(\Delta a) = \left(\frac{\Delta a}{\ell_2}\right)^{\frac{1}{2}} F(\ell_2) \quad . \tag{4-17}$$

This leads to

$$\begin{aligned} \bar{F}_c &= \left(\frac{\Delta a}{\ell_2}\right)^{\frac{1}{2}} F_e \; , \\ \bar{T}_c &= \left(\frac{\Delta a}{\ell_2}\right)^{\frac{1}{2}} T_e \; . \end{aligned} \tag{4-18}$$

The forces, F_e and T_e, at the crack tip are obtained from the coupling stiffness of the spring which is originally assumed in holding the nodes e and f together at the crack tip.

Equations (4-18) are then substituted into Equations (4-15) to calculate G_I and G_{II}. These are then substituted into Equations (4-13) to calculate K_I and K_{II}.

4.2.1.4.4 J-Integral Method

The J integral, as introduced by Rice(87), appears to offer four advantages for calculating K:

a. It can be related directly to K, in the elastic range, through Equations (4-13) since J reduces to G.

b. It can be related to crack opening displacement.

c. It can be shown to be path independent in the elastic range.

d. It has the capability to incorporate plasticity effects and appears to retain path independence into the plastic range.

These four attributes have led Verette and Wilhem[88] to propose the J integral for residual strength analysis of skin stringer when plane stress fracture dominates. A complete development follows:

The J integral is defined as

$$J = \int_{\Gamma} \{ W(\varepsilon)\, dy - \vec{T} \cdot \frac{\partial \vec{u}}{\partial x}\, ds \} \tag{4-19}$$

where W, the strain energy density, is

$$W = \int [\sigma_x d\varepsilon_x + \tau_{xy}\, d\gamma_{xy} + \tau_{xz}\, d\gamma_{xz} + \sigma_y\, d\varepsilon_y + \tau_{yz}\, d\gamma_{yz} + \sigma_z\, d\varepsilon_z \tag{4-20}$$

For generalized plane stress conditions:

$$W = \int [\sigma_x\, d\varepsilon_x + \tau_{xy}\, d\gamma_{xy} + \sigma_y\, d\varepsilon_y] \tag{4-21}$$

The contour integral J is evaluated along the curve Γ which is, in principle, any curve surrounding the crack tip. The positive direction of s in traversing Γ is counterclockwise.

Since the value of J is independent of the particular Γ contour selected, one has complete freedom in the Γ contour actually used. It appears that the path independency is maintained regardless of whether the material obeys linear elastic - nonlinear elastic - deformation theory plastic - or Prandtl-Reuss plastic constitutive relations (see Reference 56).

For ease in evaluation of J, the curve Γ can be taken to be a rectangular path (see Figure 4-18). Then dy will be nonzero only for those portions of which parallel the Y axis. Thus, since W need be evaluated only for those portions of Γ for which dy is nonzero, the computation of J is simplified.

In Equation (4-19), the second integral involves the scalar product of the tractive stress vector $\vec{T}$ and the vector whose components are the rates of change of displacement with respect to x. Resolving into components, one has

$$\frac{\partial \vec{u}}{\partial x} = \frac{\partial u}{\partial x}\,\hat{i} + \frac{\partial v}{\partial x}\,\hat{j} \tag{4-22}$$

where u and v are the displacements in the x and y directions, respectively and $\hat{i}$ and $\hat{j}$ are the corresponding unit vectors. Also

$$\vec{T} = T_x\,\hat{i} + T_y\,\hat{j} = T_1\,\hat{i} + T_2\,\hat{j} \tag{4-23}$$

where T_1 and T_2 are related to the tractive stress components through the outward normal by $T_i = \sigma_{ij}n_j$. To establish the precise form of T_i at all points along a rectangular Γ contour, consider again the crack and the surrounding Γ contour shown in Figure 4.18. An outward-pointing unit normal vector $\hat{n}$ will have components n_1 and n_2 (in the x and y directions respectively) as listed in Column (3) of Table 4-1 for the five segments of the Γ curve indicated. Applying $T_i = \sigma_{ij}n_j$, the values of T_i are given in Column (4) of Table 4-1.

The formation of the $\vec{T}$ and $\frac{\partial \vec{u}}{\partial x}$ vectors is shown in Columns (5) and (6), respectively, and the scalar product is calculated in Column (7). The relationship between ds and either dx or dy is indicated in Column (8) and the net contribution of each segment to the J integral is indicated in Column (9) of Table 4-1.

In the case of uniaxial loading, by virtue of the symmetry which exists with respect to the crack plane, one can write

$$J = 2 \int_{(x,y)_6}^{(x,y)_7} [W - \sigma_x (\frac{\partial u}{\partial x}) - \tau_{xy} (\frac{\partial v}{\partial x})]\, dy + 2 \int_{(x,y)_5}^{(x,y)_6} [\tau_{xy} (\frac{\partial u}{\partial x}) + \sigma_y (\frac{\partial v}{\partial x})]\, dx + 2 \int_{(x,y)_4}^{(x,y)_5} [W - \sigma_x (\frac{\partial u}{\partial x}) - \tau_{xy} (\frac{\partial v}{\partial x})]\, dy \qquad (4-24)$$

The J integral can be evaluated by performing the integrations indicated in Equation (4-24). The strain energy density W appearing in Equation (4-24) is, for plane stress conditions, given by Equation (4-21). In order to carry out the integration indicated in Equation (4-21), one needs a relationship between stresses and strains which realistically models the behavior actually exhibited by plastically deforming materials. For many materials, the Prandtl-Teuss equations provide a satisfactory relationship. They are, for the case of plane stress

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)
SEGMENT OF Γ CURVE	TRACTIVE STRESS COMPONENTS	COMPONENTS OF OUTWARD UNIT NORMAL	$T_i = \sigma_{ij} n_j$	$\vec{T} = T_1 \hat{i} + T_2 \hat{j}$	$\frac{\partial \vec{u}}{\partial x}$	$\vec{T} \cdot \frac{\partial \vec{u}}{\partial x}$	ds	CONTRIBUTION TO J INTEGRAL FROM SEGMENT
0-1	$\sigma_x = \sigma_{11}$ $\tau_{xy} = \sigma_{12}$	$n_1 = -1$ $n_2 = 0$	$T_1 = -\sigma_x$ $T_2 = -\tau_{xy}$	$-\left[\sigma_x \hat{i} + \tau_{xy} \hat{j}\right]$	$(\partial u/\partial x)\hat{i} + (\partial v/\partial x)\hat{j}$	$-\left[\sigma_x(\partial u/\partial x) + \tau_{xy}(\partial v/\partial x)\right]$	$-dy$	$\int_{(x,y)_0}^{(x,y)_1}\left[W - \sigma_x \frac{\partial u}{\partial x} - \tau_{xy}\frac{\partial v}{\partial x}\right]dy$
1-2	$\sigma_y = \sigma_{22}$ $\tau_{xy} = \sigma_{12}$	$n_1 = 0$ $n_2 = -1$	$T_1 = -\tau_{xy}$ $T_2 = -\sigma_y$	$-\left[\tau_{xy} \hat{i} + \sigma_y \hat{j}\right]$	$(\partial u/\partial x)\hat{i} + (\partial v/\partial x)\hat{j}$	$-\left[\tau_{xy}(\partial u/\partial x) + \sigma_y(\partial v/\partial x)\right]$	dx	$\int_{(x,y)_1}^{(x,y)_2}\left[\tau_{xy}\frac{\partial u}{\partial x} + \sigma_y \frac{\partial v}{\partial x}\right]dx$
2-5	$\sigma_x = \sigma_{11}$ $\tau_{xy} = \sigma_{12}$	$n_1 = 1$ $n_2 = 0$	$T_1 = \sigma_x$ $T_2 = \tau_{xy}$	$\sigma_x \hat{i} + \tau_{xy} \hat{j}$	$(\partial u/\partial x)\hat{i} + (\partial v/\partial x)\hat{j}$	$\sigma_x(\partial u/\partial x) + \tau_{xy}(\partial v/\partial x)$	dy	$\int_{(x,y)_2}^{(x,y)_5}\left[W - \sigma_x \frac{\partial u}{\partial x} - \tau_{xy}\frac{\partial v}{\partial x}\right]dy$
5-6	$\sigma_y = \sigma_{22}$ $\tau_{xy} = \sigma_{12}$	$n_1 = 0$ $n_2 = 1$	$T_1 = \tau_{xy}$ $T_2 = \sigma_y$	$\tau_{xy} \hat{i} + \sigma_y \hat{j}$	$(\partial u/\partial x)\hat{i} + (\partial v/\partial x)\hat{j}$	$\tau_{xy}(\partial u/\partial x) + \sigma_y(\partial v/\partial x)$	$-dx$	$\int_{(x,y)_5}^{(x,y)_6}\left[\tau_{xy}\frac{\partial u}{\partial x} + \sigma_y \frac{\partial v}{\partial x}\right]dx$
6-7	$\sigma_x = \sigma_{11}$ $\tau_{xy} = \sigma_{12}$	$n_1 = -1$ $n_2 = 0$	$T_1 = -\sigma_x$ $T_2 = -\tau_{xy}$	$-\left[\sigma_x \hat{i} + \tau_{xy} \hat{j}\right]$	$(\partial u/\partial x)\hat{i} + (\partial v/\partial x)\hat{j}$	$-\left[\sigma_x(\partial u/\partial x) + \tau_{xy}(\partial v/\partial x)\right]$	$-dy$	$\int_{(x,y)_6}^{(x,y)_7}\left[W - \sigma_x \frac{\partial u}{\partial x} - \tau_{xy}\frac{\partial v}{\partial x}\right]dy$

TABLE 4-1 TERMS USED IN J INTEGRAL DETERMINATION

$$\begin{bmatrix} d\varepsilon_x \\ d\varepsilon_y \\ d\varepsilon_z \\ d\gamma_{xy} \end{bmatrix} = \begin{bmatrix} \frac{1}{E} & -\frac{\nu}{E} & 0 & \frac{3\sigma'_x}{2\bar{\sigma}} \\ -\frac{\nu}{E} & \frac{1}{E} & 0 & \frac{3\sigma'_y}{2\bar{\sigma}} \\ -\frac{\nu}{E} & -\frac{\nu}{E} & 0 & \frac{3\sigma'_z}{2\bar{\sigma}} \\ 0 & 0 & \frac{2(1+\nu)}{E} & \frac{3\tau_{xy}}{\bar{\sigma}} \end{bmatrix} \begin{bmatrix} d\sigma_x \\ d\sigma_y \\ d\tau_{xy} \\ d\bar{\varepsilon}_p \end{bmatrix} \qquad (4\text{-}25)$$

where

$$\sigma'_x = \frac{1}{3}(2\sigma_x - \sigma_y)$$

$$\sigma'_y = \frac{1}{3}(2\sigma_y - \sigma_x) \qquad (4\text{-}26)$$

$$\sigma'_z = -\frac{1}{3}(\sigma_x + \sigma_y)$$

$$\bar{\sigma} = \left[\sigma_x^{\,2} - \sigma_x\sigma_y + \sigma_y^{\,2} + 3\tau_{xy}^{\,2}\right]^{\frac{1}{2}}$$

The primed quantities in Equations (4-26) are sometimes referred to as the deviatoric stress components in the plasticity literature. The barred quantities (i.e., $\bar{\sigma}$ and $\bar{\varepsilon}_p$) are the equivalent stress and the equivalent plastic strain.

Substituting Equations (4-25) into Equation (4-21), one obtains

$$W = \frac{1}{2E}\left[\sigma_x + \sigma_y\right]^2 + \frac{1+\nu}{E}\left[(\tau_{xy})^2 - \sigma_x\sigma_y\right] + \int \bar{\sigma}\,d\bar{\varepsilon}_p \qquad (4\text{-}27)$$

In studying Equation (4-27), observe that W will have a unique value only if unloading (at every point of the body being considered) is not permitted. To illustrate this point, consider a body that is initially unloaded and unstrained. Then

$$\sigma_x = \sigma_y = \tau_{xy} = \bar{\sigma} = \bar{\varepsilon}_p = 0.$$

If loading is applied and increased to the point where the onset of plastic action is imminent, in general σ_x, σ_y, τ_{xy}, and $\bar{\sigma}$ will be nonzero. However, the integral in Equation (4-27) will still be zero since $\bar{\varepsilon}_p$ has remained at its initial zero value. If the body were unloaded at this point, W would be a unique function of stress, regardless of loading history.

If, instead of unloading at the onset of plastic action the body is loaded into the plastic range, the integral in Equation (4-27) makes a contribution to the value of W. When the body is subsequently unloaded, the values of σ_x, σ_y, τ_{xy}, and $\bar{\sigma}$ all return to their initial zero values, but the plastic strain $\bar{\varepsilon}_p$, being unrecoverable, retains its peak value. Thus the integral $\int_o^{\bar{\varepsilon}_p} \bar{\sigma} d\bar{\varepsilon}_p$ makes a nonzero contribution to W when the body is back in its initial unloaded state. If loading into the plastic range followed by unloading is permitted W becomes multivalued. It follows that J is also multivalued for this occurrence.

The statements made in the preceding paragraph would appear to seriously limit the use of J as a fracture criterion since the case of loading into the plastic range followed by unloading (i.e., the case for which J is multivalued) occurs when crack extension takes place. On the basis of a number of examples, Hayes (Reference 56) deduced that monotonic loading conditions prevail throughout a cracked body under steadily increasing load applied to the boundaries, provided that crack extension does not occur. Thus, valid J calculations can be performed for this case.

4.2.1.4.5 Cracked Element Methods

This approach involves the use of a hybrid-cracked element which is incorporated into a finite-element structural analysis program. To date, only two dimensional crack problems can be solved with the cracked-element approach. Elements have been developed[53-55, 57-59] that allow a stress singularity to occur at the crack tip.

The cracked element consists of boundary nodal points around the geometrical boundary of the element. The element is either contained within the complete finite-element model or is solved separately using the results of finite-element analysis. In either case, the crack surface is simulated by unzipping a double-noded line along the line of expected crack extension. This builds into the structural model the proper stiffness due to the presence of the crack. The variation of stress-intensity factors (K_I and K_{II}) with crack length is determined by progressively unzipping the sets of coupled nodes.

Studies have been conducted on the variation of stress-intensity factors with cracked-element size and location[57,58]. These results define some definite guidelines in using cracked-element models. FIRST, THE DISTANCE FROM THE CRACK TIP TO THE CRACKED-ELEMENT NODAL POINTS SHOULD BE AS CONSTANT AS POSSIBLE. SECONDLY, FOR LONG EDGE-CRACKS OR CRACKS EMANATING FROM HOLES, THE CRACKED ELEMENT SHOULD ONLY CONTAIN AN AREA VERY NEAR THE CRACK TIP.

4.2.1.4.6 Strain Energy Release Rate Method

The final method uses the relation between K and the energy-release rate, G, which is defined as the derivative of the elastic-energy content of the system with respect to crack size:

$$G = dU/da. \qquad (4\text{-}28)$$

The stress-intensity factor follows from

$$K = \sqrt{E\left(\frac{dU}{da}\right)} . \qquad (4\text{-}29)$$

As in the compliance method, the elastic energy, U, can be calculated for a range of crack sizes either by solving the problem for different crack sizes or by unzipping nodes. The same advantages and disadvantages apply as to the previous method.

4.2.1.5 Comparison of Methods

Several analysis methods can be applied in deriving the variation of the stress intensity factor with crack size. Each method applied to a given problem can result in slightly different answers. As an example, several methods are applied to the problem of two cracks emanating at the edge of an open hole in a plate under tension. Two finite-element grid systems were generated to determine the sensitivity of grid size on the application of indirect methods. The following discusses the procedures used in applying each method and the comparison of results of the methods utilized.

The problem analyzed consists of a 6-inch by 12-inch plate containing a 0.5-inch diameter hole in the center.

The tensile stress is in the longitudinal direction; the crack is in the plane of symmetry and runs in transverse direction. The two grid systems consist of a fine-grid and a coarse-grid system of nodes and elements surrounding the hole. The fine-grid system has twice as many elements in the area of the hole as the coarse-grid system.

The following analysis techniques and methods were applied to each of the finite-element models to determine variation of the elastic stress-intensity factor with crack size:

a. Crack-opening displacements
b. Internal strain-energy release rate
c. Work-energy crack closure concept
d. Cracked element
e. Elastic stress field adjacent to the crack tip

f. Stress-free crack surface approach

g. Continuum mechanics solutions with secant finite-width correction.

The comparative results of the application of each method are shown in Figures 4.19 and 4.20 for the fine and coarse finite-element grid systems. As shown by the comparative plots of each method, there is scatter in the variation of the stress-intensity factor with crack size. Certain results can be rationalized as being not as accurate as others. The following discusses the application and results of each method.

a. Crack-Opening Displacement. The crack-opening-displacement (COD) results were generated based on the opening mode displacements of the nodal points of the element adjacent to the crack tip. These results show a smooth behavior for the coarse-grid model and are sensitive to grid spacing for the fine-grid model. This method appears to provide reasonable solutions for the variation of K_I with crack size for moderate size finite-element grid systems.

b. Internal Strain-Energy Release Rate. The differential of boundary force work and change of internal strain energy provides a means of determining the variation of K_I with crack size for each size of grid system. As seen in comparing Figures 4.19 and 4.20, the effect of grid system on K_I is only slight when applying the change of internal energy with crack extension ($\Delta U/\Delta a$).

c. Work-Energy Crack-Closure Concept. The work-energy method as applied to the two grid system models provided the most consistent set of results. The variation of stress-intensity-factor level with crack size demonstrated the same behavior independent of grid size. In addition, the results based on work-energy agreed with those as derived by the internal strain energy-release rate.

d. Cracked Element. The cracked element was applied to this problem in a special way. Based on previous usage of the special cracked element, the cracked element was made up of only the localized elements surrounding the crack tip. For long crack lengths, this approach was also used. Nodal-point displacements of each surrounding element as determined from the finite-element analysis are the input-boundary conditions. As seen in Figures 4.19 and 4.20, the results of the cracked element demonstrated quite a variation from the other methods at long crack lengths.

e. Elastic Stress Field. The stresses that occur at the midpoint of the element adjacent to the crack can be used to calculate the stress-intensity level for that crack according to the stress field equations of Section 4.2.1.1. The results of this approach are plotted in Figures 4.19 and 4.20. These results demonstrated very high stress-intensity levels for short cracks and not a very good comparison with the energy methods or the cracked-element results. This is due to the application of constant strain triangles on the finite-element models and analyses.

f. Stress-Free Crack Surface Approach Using Superposition. The linear superposition technique was applied to this problem. The uncracked σ_y stress distribution along the hypothetical crack line was determined for both grid systems and is shown in Figure 4.21. The method employed a weight function technique to determine the variation of stress-intensity level with crack size for a flawed hole with arbitrary crack surface pressure[48]. The crack face pressure was specified as the uncracked stress distribution of Figure 4.21. A polynomial representation of the stress distribution is used in applying the superposition technique. The results of this method are shown in Figures 4.19 and 4.20. The results are slightly higher than the results of the other methods. Again the usage of constant strain triangles in the finite-element analysis could produce stresses in slight error.

g. Continuum Mechanics Solutions. Two methods are reported in the literature for cracks emanating at open holes. The most popular solution is the one derived by Bowie[63]. This solution with the secant finite width correction factor has been plotted in Figures 4.19 and 4.20 for comparison purposes. This solution agrees with the energy solutions. Tweed and

Rooke[88] argue that the Bowie solution is in slight error for short crack lengths.

As demonstrated above, several methods are available for determining the variation of elastic stress-intensity factor with crack size using finite-element models. These results demonstrated quite a difference of K_I results depending on the method used. Therefore, extreme care should be taken when applying finite-element structural analysis programs and methods to fracture-mechanics analysis. The work-energy method seems to give the most consistent set of results and is independent of grid system size. In addition, the method also provides an average behavior when compared to the other methods previously described.

4.2.2 Stress Intensity Factors for Practical Geometries

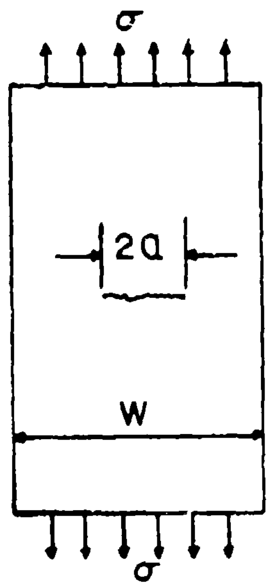
The following sections will present a catalog of available solutions for relatively simple flow geometries in uniformly loaded plates. In keeping with the philosophy presented in Section 4.2.1.1, these solutions will be presented in the form of the geometric correction factor, β. The stress intensity factor can then be obtained by direct substitution in Equation (4-5).

$$K = \beta\sigma\sqrt{\pi a} \tag{4-5}$$

Many of these solutions, while trivial in themselves, are valuable for developing more complex solutions through the method of superposition. Others are useful for obtaining approximate solutions for local effects.

4.2.2.1 Through Cracks

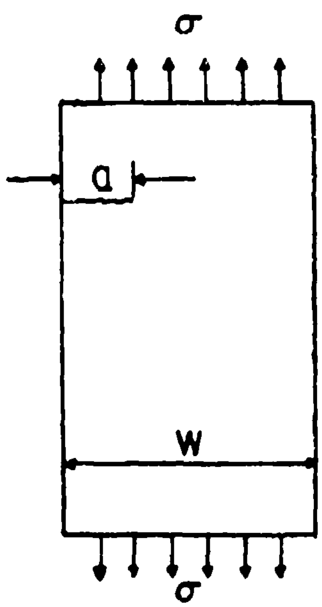
4.2.2.1.1 Finite Width Plate Under Uniform Tension (6)



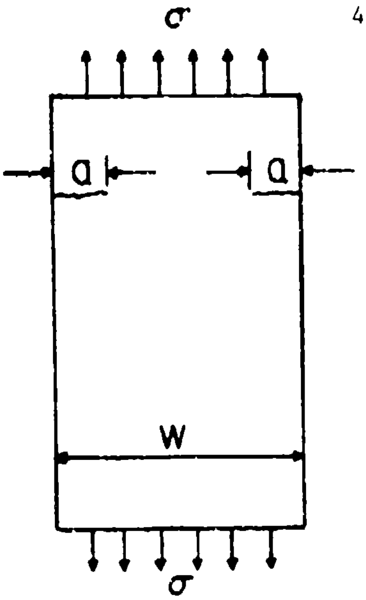
$$\beta = \left[1 + 0.256\,\frac{a}{W} - 1.354\left(\frac{a}{W}\right)^2 + 12.19\left(\frac{a}{W}\right)^3\right]$$

$$\beta \simeq \sqrt{\sec\frac{\pi a}{W}} \qquad (4\text{-}30)$$

4.2.2.1.2 Single Edge Crack Under Uniform Tension (6)

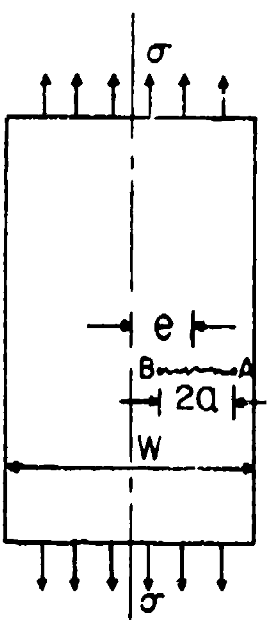


$$\beta = 1.12 - 0.23\,\frac{a}{W} + 10.6\left(\frac{a}{W}\right)^2 - 21.9\left(\frac{a}{W}\right)^3 + 30.3\left(\frac{a}{W}\right)^4 \qquad (4\text{-}31)$$



4.2.2.1.3 Double Edge Crack Under Uniform Tension (6)

$$\beta = 1.12 + 0.43\,\frac{a}{W} - 4.79\left(\frac{a}{W}\right)^2 + 15.4\left(\frac{a}{W}\right)^3 \qquad (4\text{-}32)$$



4.2.2.1.4 Eccentric Crack Under Uniform Tension (89)

$$\beta_{A,B} = F_{A,B}\,(\varepsilon,\lambda) \qquad (4\text{-}33)$$

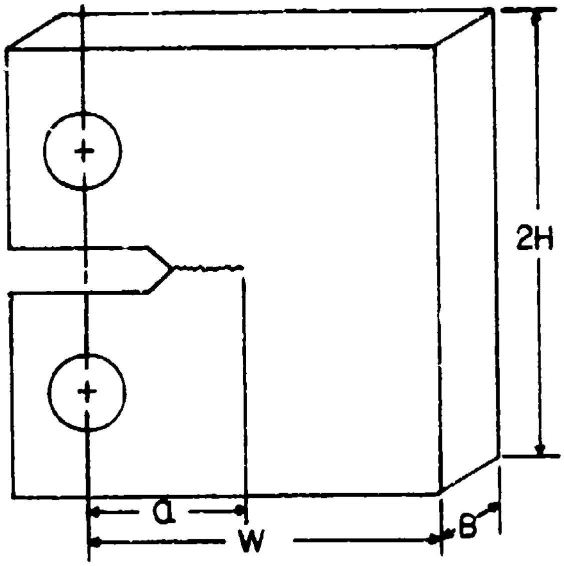
where

$\varepsilon = 2e/W$

$\lambda = 2a/(W - 2e)$

and F_A and F_B are given in Figures 4-22 and 4-23 respectively.

4.2.2.1.5 Compact Tension Specimen(90)



From the ASTM Standard E399

$$K = \frac{P}{BW^{1/2}} \left[29.6\left(\frac{a}{W}\right)^{1/2} - 185.5\left(\frac{a}{W}\right)^{3/2} + 655.7\left(\frac{a}{W}\right)^{5/2} - 1017.0\left(\frac{a}{W}\right)^{7/2} + 638.9\left(\frac{a}{W}\right)^{9/2} \right] \tag{4-34}$$

However, this equation can be put in the same form as Equation (4-5) by factoring $\sqrt{\pi a/W}$ out of the bracket. Then β becomes

$$\beta = \left[16.7 - 104.7\left(\frac{a}{W}\right)^{3} + 369.9\left(\frac{a}{W}\right)^{5} - 573.8\left(\frac{a}{W}\right)^{7} + 360.5\left(\frac{a}{W}\right)^{9} \right] \tag{4-35}$$

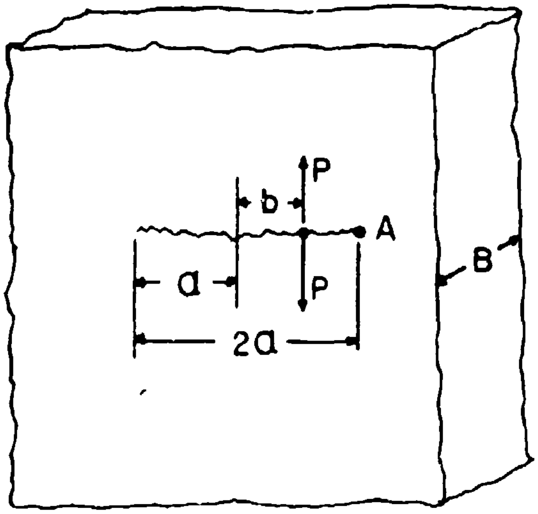
and

$$\sigma = \frac{P}{BW}$$

Equations (4-34) and (4-35) are valid for the range

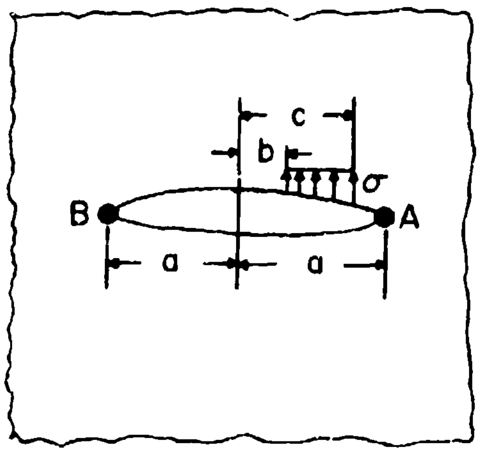
$$0.3 \le \frac{a}{W} \le 0.7$$

4.2.2.1.6 Concentrated Force on a Crack Surface(2)



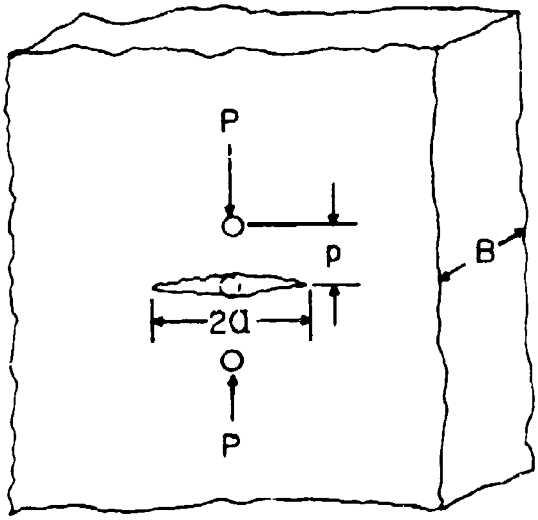
$$K_A = \frac{P}{2B\sqrt{\pi a}} \sqrt{\frac{a + b}{a - b}} \qquad (4\text{-}36)$$

4.2.2.1.7 Uniform Load on a Crack Surface(2)



$$K_A = \frac{\sigma\sqrt{a/\pi}}{2} \left\{ \sin^{-1}\frac{c}{a} - \sin^{-1}\frac{b}{a} - \left(1 - \frac{c^2}{a^2}\right)^{1/2} + \left(1 - \frac{b^2}{a^2}\right)^{1/2} \right\} \qquad (4\text{-}37)$$

4.2.2.1.8 Concentrated Force Near a Crack (91)



$$K_I = \frac{-P\sqrt{\pi a}}{2\pi B} \frac{(3 + \nu)\ p^2 + 2a^2}{(a^2 + p^2)^{3/2}} \qquad (4\text{-}38)$$

where ν = Poisson's Ratio

4.2.2.2 Part-Through Cracks

In practice, cracks usually start as semielliptical surface flaws or as quarter-elliptical surface flaws. Therefore stress-intensity solutions are required to deal with such flaw geometries because the damage assumptions in MIL-A-83444 concern elliptical flaws.

Elliptical flaws, corner flaws, and corner flaws at holes are difficult to analyze because the problem is three-dimensional in nature. THE SOLUTIONS PRESENTED ARE APPROXIMATE SOLUTIONS. DIFFERENT SOLUTIONS OF THE SAME PROBLEMS MAY SHOW SIGNIFICANTLY DIFFERENT RESULTS. In judging the effect of these differences on the predicted residual strength, it should be kept in mind that crack length

$$a = \pi\left(\frac{K}{\sigma\beta}\right)^2$$

and the residual strength for any crack length is

$$\sigma_{cr} = K_{Ic}/\beta\sqrt{\pi a}$$

Since K_{Ic} is a material constant, the relative difference in predicted residual strength for any given K solutions will be equal to the relative difference in the K solutions.

Almost all K solutions for part-through cracks, whether surface flaws or corner flaws are based on Irwin's solution[60] for a flat elliptical flaw in a plate in tension.

$$K = \frac{1.1\sigma\sqrt{\pi a}}{\left[\phi^2 - 0.212\left(\frac{\sigma}{\sigma_{ys}}\right)^2\right]^{\frac{1}{2}}} \left(\frac{a^2}{c^2}\sin^2\theta + \cos^2\theta\right)^{\frac{1}{4}} \qquad (4\text{-}39)$$

where ϕ is the elliptical integral of the second kind,

$$\phi = \int_0^{\pi/2} (1-K^2 \sin^2\phi)^{\frac{1}{2}}\, d\phi \qquad \text{with } K^2 = 1 - \frac{a^2}{c^2} ,$$

and the factor, 1.1, is an assumed correction for the front free surface. Irwin's derivation assumes that the major axis of the ellipse lies along the free surface which requires a/c 1. It is obvious from Equation (4-39) that a semi-circular surface flaw ($a/c = 1$) has a constant stress intensity factor around the crack front. The maximum stress intensity occurs at the end of the minor axis ($\theta = 0$) and the minimum occurs at the end of the major axis ($\theta = 90°$).

For convenience, define a flaw shape parameter, Q, as

$$Q = \Phi^2 - 0.212\ (\sigma/\sigma_{ys})^2$$

This reduces Equation (4-39) to

$$K = 1.1\ \sigma\sqrt{\frac{\pi a}{Q}}\ \left[\left(\frac{a}{c}\right)^2 \sin^2\theta + \cos^2\theta\right]^{\frac{1}{4}} \qquad (4\text{-}40)$$

Figure 4-24 presents Q as a function of a/2c and σ/σ_{ys}. Equation (4-40) will form the basis for all the part-through crack stress intensity factors in the same manner that Equation (4-6) is basic to all through crack stress intensity factors. It is also possible to derive a β such that Equation (4-40) may be written in the form of Equation (4-5). It is more common, however, to use the form

$$K = \sigma\sqrt{\frac{\pi a}{Q}}\ M_i\ \left[\left(\frac{a}{c}\right)^2 \sin^2\theta + \cos^2\theta\right]^{\frac{1}{4}} \qquad (4\text{-}41)$$

where M_i are various correction factors for geometry and loadings. These correction factors multiply in the same manner as the β factors discussed in Section 4.2.1.1.

The following section presents a series of correction factors, M, for various geometrical considerations.

4.2.2.2.1 Front Free Surface Correction(92)

$$M_F = 1.0 + 0.12 \left(1 - \frac{a}{2c}\right)^2 \qquad (4\text{-}42)$$

4.2.2.2.2 Back Free Surface Correction(61)

$$M_K = M_K \ (a/B,\ a2c) \qquad (4\text{-}43)$$

See Figure 4.25.

4.2.2.2.3 Surface Flaw in a Finite Plate Under Uniform Tension

Using the principle described in Section 4.2.1.1, the complete solution then becomes

$$K = \beta M_F M_K \sigma \sqrt{\frac{\pi a}{Q}} \left[\left(\frac{a}{c}\right)^2 \sin^2 \theta + \cos^2 \theta\right]^{1/4}$$

or finally

$$K = \sqrt{\mathrm{SEC}\ \frac{\pi c}{W}} \left[1 + .12 \left(1 - \frac{a}{2c}\right)^2\right] M_K \sigma \sqrt{\frac{\pi a}{Q}} \left[\left(\frac{a}{c}\right)^2 \sin^2 \theta + \cos^2 \theta\right]^{1/4} \qquad (4\text{-}44)$$

4.2.2.2.4 Corner Cracks

A corner crack can be considered as a quarter-elliptical crack. Hence, Equation (4-41) applies to a corner crack as well. Because of the two free edges being at 90 degrees (instead of at 180 degrees, as in the case of a surface flaw), the front-free-surface correction must be modified. Liu[69] has developed an approximate solution for a quarter circular flaw in a quarter infinite solid based on Smith's[93] solution for semi-circular cracks. For a semi-circular flaw in a semi-infinite solid Equation (4-41) becomes

$$K = 2\sigma\sqrt{a/\pi}\ \ M_F$$

However, the corner flaw has two free edges so the expression becomes

$$K = 2\sigma\sqrt{a/\pi} \cdot M_F\ (0°) \cdot M_F\ (90°)$$

From Smith's solution M_F (0°) = 1.03 and M_F (90°) = 1.22. Hence, Liu's solution becomes

$$K \simeq \sigma\sqrt{2a} \qquad (4\text{-}45)$$

Kobayashy and Enetanya[62] have calculated more precise stress-intensity factors for elliptical corner cracks. They arrive at

$$K = M_K F_K \frac{\sigma}{\phi}\sqrt{a} \left(\frac{a^2}{c^2} \sin^2 \theta + \cos^2 \theta\right)^{1/4} \qquad (4\text{-}46)$$

where F_K is as given in Figure 4.26. F_K is maximum near the edges (surface) and it is on the order of 1.3 rather than 1.2.

4.2.2.3 Cracks at Holes

4.2.2.3.1 Radial Through Crack at Open Hole Under Uniform Tension

On the basis of the work by Bowie[63], the stress-intensity factor for a through crack at a hole in an infinite plate (Figure 4.27) is given by

$$K = \sigma\sqrt{\pi a}\;\; M_B\left(\frac{a}{D}\right) \tag{4-47}$$

where a is the size of the crack as measured from the edge of the hole and D is the hole diameter. The function M_B (a/D) can be given in tabular or graphical form as M_{B1} for a single crack and M_{B2} for the symmetric case with two cracks. Grandt[48] has recently developed a least-squares fit to M_B of the form

$$M_B\,(a/D) = \frac{C_1}{C_2 + a/D} + C_3 \tag{4-48}$$

where C_1, C_2, and C_3 have values as given in Figure 4.27.

Tweed and Rooke[88] have improved upon the accuracy of the Bowie solution, particularly in the small crack region. Brussat[94] developed the following curve fit to the numerical solution developed by Tweed and Rooke which agrees within one percent for any value of a/R.

$$M_B = \mathrm{EXP}\left[1.2133 - 2.205\left(\frac{a}{a+R}\right) + .6451\left(\frac{a}{a+R}\right)^2\right] \tag{4-49}$$

The Tweed and Rooke solution is only for the asymetric case presented in

Figure 4.27. Comparison of Equations (4-48) and (4-49) reveals a maximum of difference of two percent over a range of a/R from 0.01 to 10.

4.2.2.3.2 Radial Through Cracks at Pin Loaded Holes

Shah has developed approximate solutions for loaded holes in infinite plates based on a Green's function approach. For radial through cracks, the solutions are presented in Figures 4.28 and 4.29. The solutions are presented as a function of the pin load for both single and double cracks. The expression is

$$K = \sigma_b \sqrt{\pi a} \;\; M_{1,2} \; (a/D) \qquad (4\text{-}50)$$

where

$\sigma_b = P/Dt$ is the bearing stress

$M_1(a/D)$ = Figure 4.28

$M_2(a/D)$ = Figure 4.29

4.2.2.3.3 Part Through Cracks at Open Holes Under Uniform Tension(70)

Shah has also developed approximate K solutions for part through cracks using the Green's function approach. Non-dimensional correction factors were obtained for the double crack cases in infinite plates. A modification factor to correct these solutions to single crack cases was also developed. From Equation (4-41) Shah's solution may be written for finite plates as shown:

Double Crack

$$K_{2H} = M_F\ (a/C) \cdot M_B\ (a/c,\ a/t,\ \theta) \cdot M_{2H}\ (C/D,\ \theta) \cdot K_{Ie} \qquad (4\text{-}51)$$

where K_{Ie} = Figure 4-31

M_F (a/c) = Equation 4-42

M_B (a/c, a/t, θ) = Figure 4-30 (other a/c values in reference 95)

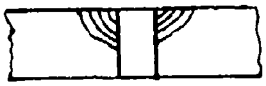
M_H (a/D, θ) = Figure 4-31

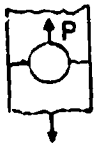
Single Crack

$$K_{1H} = K_{2H} \cdot \frac{D + \pi ac/4t}{D + \pi ac/2t} \qquad (4\text{-}52)$$

4.2.2.3.4 Part Through Cracks at Pin Loaded Holes[70]

Double Crack





$$K_{2P} = M_F\ (a/C) \cdot M_B\ (a/c,\ a/t,\ \theta) \cdot M_{2P}\ (C/D,\ \theta)\ K_{Ieb} \qquad (4\text{-}53)$$

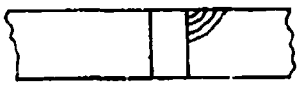
where K_{Ieb} = Figure 4-32

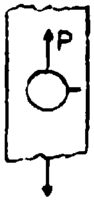
M_F (a/c) = Equation 4-42

M_B (a/c, a/t, θ) = Figure 4-30 (other a/c values in reference 95)

M_P (c/D, θ) = Figure 4-32

Single Crack





$$K_{1P} = K_{2P} \cdot \frac{D + \pi ac/4t}{D + \pi ac/2t} \qquad (4\text{-}54)$$

4.2.2.3.5 <u>Comparison of Other Corner-Crack at Hole Solutions</u>

MIL-83444 places emphasis on corner cracks emanating from holes. A rigorous solution for flawed holes does not exist. However, stress-intensity estimates have been reported(68-70) employing elliptical crack solutions and correction factors to account for the hole. For some configurations, stress-intensity factors were determined experimentally(71,72). A number of these solutions are described in subsequent paragraphs and compared with the solutions of 4.2.2.3.3.

A STRAIGHTFORWARD ENGINEERING SOLUTION IS SOMETIMES USED, BY TAKING THE STANDARD ELLIPTICAL FLAW SOLUTION AND APPLYING THE BOWIE CORRECTION FACTOR, AS IF IT WERE A THROUGH CRACK,

$$K = \sigma\sqrt{\frac{\pi a^2}{Qc}}\, f_B\left(\frac{c}{D}\right) , \qquad (4\text{-}55)$$

where Q is the flaw shape parameter shown in Figure 4.24 and f_B is the Bowie function given in Figure 4.27. For the case of a quarter-circular flaw with a = c, Equation (4-55) reduces to

$$K = \frac{2\sigma}{\pi}\sqrt{\pi a}\, f_B\left(\frac{a}{D}\right) , \qquad (4\text{-}56)$$

The equation is limited to cases where a/B < 0.5, B being the thickness, unless a back-free-surface correction would be applied.

Hall and Finger[68] derived an empirical expression on the basis of failing stresses of specimens with flawed holes, assuming the specimens failed when K reached the standard K_{Ic}. They arrived at

$$K = 0.87\, \sigma\sqrt{\pi c_e}\; f_B \left(\frac{c_e}{D}\right) . \qquad (4\text{-}57)$$

In this equation, c_e represents an effective crack size, which has to be found from the empirical curves in Figure 4.33. It incorporates the influence of both flaw shape and back-free surface, but it is limited to $a/c < 1$. The Bowie function, f_B, is also based on the effective crack size, c_e.

Liu[69] considered a quarter-circular flaw, such that the flaw shape parameter Φ equals $\pi/2$. He arbitrarily based the Bowie function on an effective crack, $a_e \approx 1/2\ (a\sqrt{2})$. His equation then is

$$K = \alpha_b\, \alpha_f \frac{\sigma}{\phi} \sqrt{\pi a}\; f_B \left(\frac{a_e}{D}\right) \qquad (4\text{-}58)$$

A corner flaw has two free surfaces, which can be accounted for by a free-surface correction of 1.2 to 1.3. Since the edge-crack-surface correction is already included in the Bowie function, Liu took the free-surface correction as $\sigma_f = 1.12$. Taking the back-free-surface correction, σ_b, equal to unity and noting the $\phi = \pi/2$ for $a = c$, the final equation is

$$K = \frac{2.24\sigma}{\pi} \sqrt{\pi a}\; f_B \left(\frac{a_e}{D}\right), \text{ with } a_e = 1/2\ (\sqrt{2}a) \qquad (4\text{-}59)$$

Hall, Shah, and Engstrom[70] also presented an analysis method for elliptical cracks emanating from holes. They used the solution for a pressurized elliptical crack with a pressure distribution in the form of a polynomial. They fitted the polynomial roughly to the stress distribution around an uncracked hole in a plate under tension. Then, they solved the problem of an elliptical crack (without a hole) with the calculated pressure distribution. The result is obtained from Equation 4.52. It is also slightly dependent on a/c, but the variations are within 6 or 7 percent.

Hall, Shah, and Engstrom checked their procedure by applying it to a through crack and found it applicable. They also showed that the case of an elliptical crack reduces to the Bowie solution for a/c approaching infinity. The stress-intensity factor is then $K = \sigma\sqrt{\pi c}\; M_{2H}$ (c/D, 90°), implying that values of M_H (c/D, 90°) in Figure 4.31 should be equal to the Bowie function f_B (c/D). In Figure 4.34, this is shown to be the case.

A comparison of solutions is made in Figure 4.34. In view of its pertinence to the MIL-A-83444 damage assumptions, only quarter-circular flaws are considered. The figure is limited to the case that $a/B < 0.5$, such that back-free-surface corrections can be neglected. This introduces a difficulty with the Hall and Finger equation in that the value of a_e is strongly dependent on the a/B ratio for $a/B < 0.5$. In view of this, a range of a/B of 0.1 to 0.4 was taken for the Hall and Finger relation, which corresponds with a range of c_e/c of 0.15 (extrapolated) to 0.7 (Figure 4.33).

Another difficulty arises because the Hall, Shah, and Engstrom analysis essentially considers the variation of K along the crack front. Therefore the K values are given for $\beta = 0°$ (edge of the hole), $\beta = 20°$, and $\beta = 90°$ (surface). As explained previously, the case of $\beta = 90°$ coincides with the Bowie case.

Hall, Shah, and Engstrom applied their analysis to a limited series of fracture toughness specimens. They calculated the stress intensity at fracture as a function of crack front angle β. They found that the stress intensity at fracture was higher than K_{Ic} for $0 < \beta < 20°$ and lower than K_{Ic} for $\beta > 20°$. Therefore, they concluded that $\beta = 20°$ is the critical point. In the region of $\beta = 20°$, the gradient of K is not large. Hence, the conclusion on what is the critical point becomes very sensitive to the K_{Ic} value chosen as representative.

The line for $\beta = 30°$ to $40°$ will come close to the other solutions. Then, the K values predicted by all solutions approach each other for flaw sizes larger than the hole diameter. The experimental data shown in Figure 4.33 were obtained from photoelastic measurements[72]. They are at least on the same order of magnitude as the predictions. It is noteworthy that Liu's solution predicts K values for small flaws almost as high as for through cracks.

Almost all analysis methods gave good results when applied to certain sets of fracture data. Equation (4-57) by Hall and Finger matched their data within 10 percent. Liu applied his Equation (4-58) to the same data and found fair agreements. The data covered fairly large values of the ratio a/D. Also the data by Hall and Engstrom were for large a/D. Their results are analyzed by means of the Hall and Finger equation under the assumption that the flaws were quarter-circular. The results are remarkably good. Since a/B for these data was rather large, the data fall near the upper boundary in Figure 4.34. This is not too close to the line for $\beta = 20°$, considered critical by Hall and Engstrom, but it would be close to a line for $\beta = 30°$. NOTE THAT RESIDUAL STRENGTH PREDICTIONS MADE WITH THE GIVEN K SOLUTIONS WOULD GIVE THE SAME SPREAD OF VALUES AS IN THE K SOLUTIONS.

4.2.2.3.6 Effects of Interference and Cold Working

When considering a crack emanating from a fastener hole, the influence of the fastener has to be taken into account. If the fastener is a loose fit in an otherwise untreated hole and when there is no load transfer, it is likely to have little effect on the behavior of a crack emanating from the hole. In general, however, the fastener has a tight (interference) fit. In many cases, it does transfer some load. Moreover, the holes are often cold worked to improve fatigue resistance. All these things have an effect on cracking behavior, since they induce a redistribution of local stresses to the effect that the stress intensity is different from that at a cracked open hold. The damage tolerance requirements in MIL-A-83444 prescribe that all these effects be accounted for by the use of smaller assumed initial damage sizes.

Application of fracture mechanics principles to cracks at filled fastener holes required knowledge of the effect of interference, cold work, and load transfer on the stress-intensity factor. Attempts to approach this problem were made by Grandt[48] and Shah[70]. Grandt calculated stress-intensity factors for cold-worked and interference-fit holes by solving the problem of a cracked hole with an internal pressure distribution equal to the hoop stress surrounding an uncracked fastener hole. Shah used a Green's function approach with approximations for the stress distributions at the hole.

Figures 4.35 and 4.36 show the observed trends. Since the shape of the curves depends upon the applied stress, the calculation must be repeated for different stresses. Consequently, the results cannot be presented nondimensionally. The results in Figure 4.35 may be slightly misleading, because the hoop stress will be partly released when the bolt gets more clearance as the crack grows (decreasing stiffness). This effect was not accounted for in Grandt's solution.

Figures 4.35 and 4.36 indicate that both an interference fit and cold work significantly affect the stress intensity. Mandrelizing is more effective, since it gives a larger reduction of the stress intensity over a wider range of a/D values. This range is particularly important for fatigue-crack growth, since the larger part of the life is spent while the cracks are still small.)

4.2.3 Crack Growth Resistance Curve Approach

Fracture mechanics based methods have shown promise due to their success in predicting residual strength in plane strain or small scale yielding problems. Attempts have been made to extend tne linear elastic fracture mechanics methods to treat large scale yielding problems. A summary of these methods is given in References 96 and 97. It was concluded that an accuracy of ± 5% was possible in obtaining the required fracture criterion data. However, the available mathematical tools (e.g., References 76 and 98) were not as accurate as the finite element method using either special cracked elements or the procedure described in Reference 79. In order to treat the problem of slow, stable tear associated with high toughness thin section fracture, the crack growth resistance curve (K_R) showed good promise (Reference 99) but difficulties in estimating crack tip plasticity have led to an alternate failure criterion.

The alternate criterion employs Rice's J integral (Reference 87) in combination with slow tear or a $\sqrt{J_R}$ versus Δa curve (for skin critical structure). This criterion (incorporating slow tear) was proposed in both References 100 and 101 for other than plane strain fracture. Its application to structural problems was proposed in References 96 and 100. The analysis involves computation of J values for the structure of interest for successive crack sizes and a tangency condition similar to the K_R curve approach. This approach has the current capability to consider the majority of structural, loading and material parameters which were given in Table I of Reference 96 (see Table VIII) for the "ideal" residual strength prediction technique. The technique as presented in reference 102 represents a major step forward in analyzing residual strength of through cracked complex structural arrangements where slow stable tear and large plastic zones prevail.

4.2.3.1 The J Integral

The J integral has been investigated by several researchers as a failure criterion for plane strain fracture (References 8, 103, and 104). In Reference 96 the suitability of the J integral as a failure criterion for plane stress fracture was described.

The J integral is defined by Rice (Reference 87) as

$$J = \int_{\Gamma} (W dy - \vec{T} \cdot \frac{\partial \vec{u}}{\partial x} ds) \qquad (4\text{-}60)$$

where Γ is any contour surrounding the crack tip, traversing in a counter clockwise direction.

W is the strain energy density

$\vec{T}$ is the traction on Γ, and

$\vec{u}$ is the displacement on an element along arc s.

The strain energy density W is given by

$$W = \int [\sigma_x \, d\varepsilon_x + \tau_{xy} \, dr_{xy} + \tau_{xz} \, dr_{xz} + \sigma_y \, d\varepsilon_y + \tau_{yz} \, dr_{yz} + \sigma_z \, d\varepsilon_z]$$

and for generalized plane stress (4-61)

$$W = \int [\sigma_x \, d\varepsilon_x + \tau_{xy} \, dr_{xy} + \sigma_y \, d\varepsilon_y] \ .$$

For elastic material behavior, J is equivalent to Irwin's strain energy release rate G. For Mode 1, the relation between G and stress intensity factor K_I is given by

$$G = \frac{1 - \nu^2}{E} K_I^2 \quad \text{for plane strain}$$

$$G = \frac{K_I^2}{E} \quad \text{for plane stress} \qquad (4\text{-}62)$$

Thus for elastic material behavior J can be related to stress intensity factor K. Contrary to K or G, the use of J is not restricted to small scale yielding. J can be used as a generalized fracture parameter even for large scale yielding, (see e.g., Reference 15). For an elastic-perfectly-plastic material (materials exhibiting Dugdale type plastic zones - see References 1 and 34), J is directly related to crack opening displacement, COD and for such a material the relationship is given by

$$\delta = \frac{J}{F_{ty}} \ .$$

4.2.3.2 J Integral as Failure Criterion

The application of the J integral as a fracture criterion has been mainly restricted to plane strain fracture and its application to plane stress fracture has not been studied in any extensive manner. This perhaps is due to slow stable tear that normally accompanies plane stress fracture. The J integral can be used to predict plane stress fracture if a J_R resistance curve similar to the recently proposed K_R resistance curve can be obtained. In practice it is perhaps desirable to plot square root of J_R ($\sqrt{J_R}$) since for elastic cases it is directly related to stress intensity factor versus crack extension. This curve will have the form shown in Figure 4.37.

In Reference 32, Kraft, et al, first introduced a failure criterion based on crack growth resistance concepts, or K_R. They suggested that a crack will grow stably if the increase in resistance as the crack grows is greater than the increase in applied stress intensity. If these conditions are not met, unstable fast fracture will occur. This fast fracture occurs when

$$K = R, \qquad \text{and} \qquad \frac{\partial K}{\partial a} \geq \frac{\partial R}{\partial a}$$

The method of employing stress intensity factors along with a K_R curve has several disadvantages, however, which are associated with estimates of crack tip plasticity. This concept can be extended to incorporate plasticity effects by using J in place of the stress intensity factor K.

The use of J has several advantages as discussed in References 96 and 100. A brief outline of the procedure involves the following steps.

- Obtain $\sqrt{J_R}$ curve for the skin material of the structure using a suitable specimen (e.g. crack line wedge loaded or center cracked tension).
- Obtain J values for the stiffened structure for various cracks lengths and applied stresses using a suitable plasticity model (e.g. Dugdale Model).
- Determine the point of instability from the $\sqrt{J}$ curves of the structure and $\sqrt{J_R}$ curves of the material as shown schematically in Figure 4.38.

The square root of J versus crack length are plotted for various applied stresses. The $\sqrt{J_R}$ resistance curve is superimposed on the diagram at some physical crack length under consideration, say a_o. The corresponding failure stress is given by the point of tangency between $\sqrt{J_R}$ curve and $\sqrt{J}$ curve at point A. Thus fracture stress is given by σ_4 in Figure 4 with associated slow tear of the amount Δa.

4.2.3.3 Residual Strength Predictions

Using the J integral as the failure criterion Ratwani and Wilhem (reference 102) developed a step-by-step procedure for predicting the residual strength of built-up skin stringer structure.

The residual strength prediction procedure is briefly outlined here to show, step-by-step, the required data and analysis. It should not be assumed that by reading this step-by-step procedure that the uninitiated can perform a residual strength prediction. It is strongly recommended that the details of all preceding sections be examined prior to attempting a structural residual strength analysis following these ten procedural steps.

STEP 1. Model the structure for finite element analysis or use an existing finite element model remembering --

a. Two dimensional structural idealizations
b. No out-of-plane bending permitted
c. Use proper fastener model, flexible fastener model for riveted or bolted structure or the shear spring model for bonded structure
d. Use material property data from skin and substructure of interest (i.e., E, F_{ty} and F_{tu})
e. Select most critical location for crack (normally highest stressed area)
f. Take advantage of structural symmetry.

Step 2. Select one crack length (2a or a) of interest (based on inspection capability or detailed damage tolerance requirement). Based on this "standard" crack length, five other crack lengths are selected for

Dugdale type elastic plastic analysis. These crack lengths should be selected such that crack length to stiffener spacing (2a) ratios vary between 0.15 to 1.1 remembering --

a. That the greatest variation is J values will take place near reinforcements

b. To select at least one crack size shorter than "standard"

STEP 3. With finite element model (from Step 1) and assumed crack lengths (from Step 2) perform analysis assuming Dugdale type plastic zones for each crack size remembering--

a. To select first increment of plastic zone length at 0.2 inches and sufficient successive increments (normally 6) to reach Buekner-Hayes calculated stresses up to 85 percent to F_{ty}.

b. Make judicious selection of plastic zone increments so as to take advantage of overlapping a_e values (e.g., 3.2, 3.5, 4.2 5.0 inches for a 3 inch physical crack and 4.2, 4.5, 5.0 inches, etc., for a 4 inch physical crack). If overlapping is done those cases where the crack surfaces are loaded thoughout the crack length will be common for two or more physical crack sizes hence the computer programs need be run only once (e.g., 4.2 and 5.0 inches) thus reducing computer run times.

STEP 4. From Step 3 obtain stresses in stiffeners for Dugdale analysis and elastic analysis. Plot stiffener stresses as function of applied stress.

STEP 5. From the crack surface displacement data of Step 3 plot $\sqrt{J}$ (obtained by Buekner-Hayes approach) versus applied stress to F_{ty} ratio for each crack size.

STEP 6. From Step 5 cross plot the data in the form of J versus crack size (a) at specific values of applied stress to F_{ty} ratio.

STEP 7. Employing the data of Step 4 and the "standard" crack size determine, gross panel stress to yield strength ratio, s/F_{ty} at ultimate strength (F_{tu}) for the stiffener material - assuming zero slow crack growth. This information will be used subsequently to determine if a skin or stiffener critical case is operative.

STEP 8. Obtain crack growth resistance data for skin material (see Volume II of reference 102) remembering--

a. To use thickness of interest (i.e., if chem milled 7075-T6 use chem milled 7075-T6 material)

b. Use proper crack orientation LT or TL or off angle to correspond to anticipated structural cracking.

STEP 9. Plot $\sqrt{J}$ versus Δa_{PHY} curve from the data obtained in Step 8.

STEP 10. Determine structural residual strength. On the $\sqrt{J}$ versus crack size (a) plots obtained in Step 6 for the structure, overlay the $\sqrt{J_R}$ versus Δa_{PHY} material plot of Step 9 at the initial crack length of interest. (This procedure is shown in the next subsection.) Determine the--

a. At the gross panel stress obtained from Step 7 significant slow tear ($\geq$ 0.25 inch) will occur as indicated from the intersection of the $\sqrt{J_R}$ versus Δa_{PHY} curve with the constant p/F_{ty} curve at a stringer ultimate strength (see Step 7). Interpolation will probably be necessary between values of constant p/F_{ty}. Then proceed as follows:

- If significant slow tear occurs ($\geq$ 0.25 inch) the structure can be considered to be skin critical (at that particular crack length). Tangency of $\sqrt{J_R}$ versus Δa_{PHY} and $\sqrt{J}$ versus a_{PHY} at constant applied stres ., be used to determine extent of slow tear and residual strength at failure as a percentage of F_{ty}.
- If significant slow tear does not occur (Δa_{PHY} < 0.25 inch) the structure will normally be stiffener critical. To determine a conservative value of residual strength (for that crack length) use the Dugdale curve of Step 4 and stiffener ultimate strength.

The most important factor to consider in residual strength prediction of a cracked built-up structure is to decide whether the structure is skin or stiffener critical. Normally a short crack length is likely to be a skin critical case and a long crack length a stiffener critical case. However there is no clear cut demarcation between the two cases. Factors such as percentage stiffening, spacing of stringers, lands in the structure and other structural details will influence the type of failure. Hence, a good technique is to determine the residual strength of a given structure based on both skin critical and stiffener critical cases. The minimum fracture stress of the two will then represent the residual strength of the structure and should be considered to be the governing case.

4.3 DATA REQUIREMENTS

4.3.1 Materials Characterization

For over a decade linear elastic fracture mechanics (LEFM) has been used successfully as a tool in studying fatigue crack propagation and brittle fracture in solids. However the role of linear elastic fracture mechanics is restricted to brittle and semi-brittle materials and low stress levels where plasticity is confined to a very small region ahead of the crack tip (i.e. the crack tip plasticity does not significantly alter the behavior of the material ahead of the crack tip). In LEFM the most widely used single factor to study fatigue and fracture phenomena is the stress intensity factor (K). For elastic material behavior, the relationship between stress intensity factor K, strain energy release rate (G) and J integral is well known (see e.g., Reference 87). Thus all elastic approaches to residual strength prediction essentially use as a failure criterion critical stress intensity (K_c) or modification to K_c to account for plasticity.

4.3.1.1 Fracture Toughness

The critical K for fracture is denoted as K_{Ic} for plane-strain conditions and K_c for plane-stress conditions. Within the limitations discussed in subsequent sections, K_{Ic} and K_c can be considered as a material property called fracture toughness (with adjectives plane strain or plane stress, respectively). They can be determined by experiment. For example, a plate of given dimensions and known crack size can be loaded in tension to fracture. The stress at fracture and the crack size can be substituted in the appropriate K equation. The value of the expression is then set equal to K_{Ic}. Subsequently, the fracture stress (or critical crack size at a given stress) can be calculated for any other crack configuration for which an expression for the stress-intensity factor is known.

4.3.1.1.1 Testing Procedures

The test procedure for plane strain fracture toughness testing is defined by the American Society for Testing and Materials (ASTM) by test method E-399. It is adequately described elsewhere (e.g., References 21 and 22). The recommended specimens contain a machined starter notch from which a fatigue crack is initiated by cyclic loading.

The cracked psecimen is subjected to a fracture test. During the test, the displacement of the crack edges [crack-opening displacement (COD)] is measured by means of a strain-gaged clip gauge[20,21]. The load and the COD are recorded on a X-Y recorder. In the ideal case, the load-COD diagram is a straight line up to the point of fracture. In that case, the fracture load is substituted in the K expression to calculate K_{Ic}. A limited nonlinearity of the load-COD diagram is accepted, provided the screening criteria for the test are met[20,21].

The most important screening criterion is the thickness criterion. After the test, it should be checked whether $B > 2.5(K_Q/\sigma_{ys})^2$, where K_Q

is the measured apparent K_{Ic}. If this (and other) criteria are satisfied, K_Q is declared a valid K_{Ic}. The thickness criterion ensures that plane strain prevailed during the test.

A generally accepted method for plane stress and transitional fracture toughness testing and presentation of results does not exist. No ASTM standard is available. In the absence of any recognized standards two approaches will be outlined in a subsequent section.

4.3.1.1.2 Plane Strain Fracture Toughness

The toughness of a material largely depends upon thickness. This is shown diagrammatically in Figure 4.39, where the thickness is nondimensionalized by dividing by $(K_{Ic}/\sigma_{ys})^2$ which is a measure of the plastic zone size. The thickness effect on toughness is associated with the state of stress at the crack tip. In thick plates, the state of stress is plane strain. The toughness in the plane strain regime is virtually independent of thickness. The plane strain fracture toughness, K_{Ic}, is indicated in Figure 4.39. For increasing thickness the toughness asymptotically approaches K_{Ic}.

The plane strain fracture toughness of a material depends strongly on yield strength, as illust ated in Figure 4.40 for different alloy systems[23-25]. Variations in toughness also occur as result of anisotropy. Usually, there are appreciable differences in toughness for different crack-growth directions. The toughness in the short transverse direction is always the lowest. For an aluminum-zinc-magnesium alloy, toughness values are reported[24] of 36 ksi $\sqrt{in.}$, 19 ksi $\sqrt{in.}$, and 15 ksi $\sqrt{in.}$ for the longitudinal, transverse, and short transverse direction, respectively.

These toughness variations arise from chemical banding and preferential orientation of impurities, due to the rolling or forging operation. Also, the shape and orientation of the grains as affected by mechanical processing can have an effect on fracture toughness. IN A RESIDUAL STRENGTH ANALYSIS OF A STRUCTURAL DESIGN, THE DIRECTION OF CRACK GROWTH SHOULD BE WELL IDENTIFIED AND APPROPRIATE FRACTURE TOUGHNESS VALUES SHOULD BE USED. THIS IS OF PARTICULAR IMPORTANCE FOR FORGINGS, SINCE THE GRAIN FLOW MAY VARY FROM PLACE TO PLACE.

Plane strain fracture toughness values for many alloys are compiled in the Damage Tolerance Design Handbook[20]. Figure 4.41 is a reproduction of a page of this Handbook, showing how the data are presented. The table also gives an indication of the scatter in K_{Ic} data. For example, the average room-temperature values of Code 1, A, B, and C (in the table) vary between 50.5 and 55.9 ksi $\sqrt{in.}$, whereas the minimum value was as low as 48.8 ksi $\sqrt{in.}$ In such a case, no values higher than 50.5 ksi $\sqrt{in.}$ are recommended for use.

4.3.1.1.3 Plane Stress and Transitional Behavior

In very thin plates, the crack tip is under plane stress. If a condition of plane stress can fully develop, the toughness reaches a maximum $K_{c(max)}$, the plane stress fracture toughness. (Usually there is a small decay in toughness for thin sheets, below the full plane stress thickness.) Thereafter, the toughness gradually decreases from $K_{c(max)}$ to K_{Ic} with increasing thickness.

The state of stress at the crack tip is at least biaxial, which is plane stress. In this case, the stress in thickness direction, σ_3, is zero. Since the other stresses are high, the strains and displacements in thickness direction are appreciable. There exists a large stress gradient at the crack tip, which means that the tendency to contraction in thickness direction differs largely for adjacent material elements. As a result, more remote elements will constrain the contraction of elements close to the crack tip. Clearly, the constraint will be larger if the required displacements are larger (i.e., if the plate is thicker). In the ultimate case the contraction is fully constrained (plane strain).

When a thin plate is loaded to K_{Ic}, the plastic zone is already on the order of the plate thickness. Plane stress develops, the plastic zone becomes large, and deformation becomes easier. Therefore, K_{Ic} is not enough for fracture. The plate can be loaded to a much higher K before it reaches a critical value that causes fracture. This plane stress fracture toughness is usually denoted as K_c. Plate of intermediate thickness become critical at K values somewhere between K_{Ic} and K_c.

The transition from plane stress fracture to plane strain fracture is associated with a change in fracture plane(11). This is also shown in Figure 4.39. In the case of plane stress, the crack propagates on a plane at 45 degrees to both the loading direction and the place surface. In the case of plane strain, the crack propagates on a plane perpendicular to both the loading direction and the plate surface. Along the edges of the crack, the fracture surface is slanted at 45 degrees where it cuts through the plane stress region at the plate surface. The slant edges are called shear lips. In the transitional region the shear lips constitute a bigger part of the total fracture surface. In the absence of a plane-strain region, the two shear lips meet to form a completely slant fracture.

Actual data, distinctly showing the behavior depicted in Figure 4.39, are scarce. The scatter in the transitional region is usually so large that a reliable curve hardly can be drawn. Some data[16] are compiled in Figure 4.42. Plates of different thickness are usually from different heats of material. As a result, their yield stress will be different. The strong dependence of toughness on yield strength then is responsible for the scatter in the data[11,16]. If the plates would be machined from the same stock, scatter likely would be less.

Various models have been proposed[16-19] to account for the thickness effect. Most of these models predict a much stronger dependence of toughness on thickness than actually observed, except for the engineering approach suggested by Anderson[16]. He proposed a linear decay of toughness between the maximum (plane stress) value and K_{Ic}.

4.3.1.1.4 Slow Stable Crack Growth

Consider a sheet or plate with a central transverse crack loaded in tension at a nominal stress, σ, as shown in Figure 4.43. The stress can be raised to a value, σ_i. Then the crack starts to propagate. Crack growth is slow and stable; it stops practically immediately when the load is kept constant. Although the crack has increased in length, a higher stress is required to maintain its growth. Finally, at a certain critical stress, σ_c, a critical crack size, a_c, is reached. Crack growth becomes unstable and a sudden total fracture results. When the initial crack is longer, crack growth starts at a lower stress. Also, the fracture stress (residual strength) is lower, but usually there is slower crack extension prior to fracture.

The slow stable crack growth is dependent upon testing system stiffness and specimen geometry. However, it will still occur in a soft testing system where no drop of load takes place when the crack propagates. Slow crack growth may be on the order of 20 to 50 percent of the initial crack size[26] depending upon alloy type and testing conditions.

It can be assumed that all events described in the foregoing paragraphs are governed by a critical stress-intensity factor. Each event can be labeled by a stress-intensity expression, i.e.,

$$
\begin{aligned}
&\text{Stress intensity at onset of crack growth,} && K_i = \beta\sigma_i\sqrt{\pi a_i} \\
&\text{Stress intensity at instability,} && K_c = \beta\sigma_c\sqrt{\pi a_c} \qquad (4\text{-}63)\\
&\text{Apparent stress intensity at instability,} && K_{app} = \beta\sigma_c\sqrt{\pi a_i}\ .
\end{aligned}
$$

A crack of size a_i starts propagating slowly at σ_i, which means that the critical stress intensity for the onset of slow growth is K_i as defined above. Similarly, K_c is the critical stress intensity for instability.

K_{app} IS AN APPARENT STRESS INTENSITY; THE EXPRESSION COMBINES THE CRITICAL STRESS AND THE INITIAL CRACK SIZE WHICH DO NOT OCCUR SIMULTANEOUSLY. HOWEVER, IT IS THE INITIAL FATIGUE CRACK SIZE THAT IS ASSOCIATED WITH THE GIVEN RESIDUAL STRENGTH. FROM A TECHNICAL POINT OF VIEW, IT IS IMMATERIAL WHETHER THIS CRACK SHOWS STABLE GROWTH BEFORE FRACTURE. WHAT MATTERS IS ITS FRACTURE STRESS. HENCE, K_{app} DOES HAVE TECHNICAL SIGNIFICANCE. IT CAN BE USED TO CALCULATE THE RESIDUAL STRENGTH: I.E., THE STRESS AT WHICH A CRACK OF A GIVEN INITIAL SIZE BECOMES UNSTABLE.

Tests have shown that K_i, K_c, and K_{app} are not constants with general validity like K_{Ic}. But to a first approximation, they are constant for a given thickness and for a limited range of crack sizes. For a given material with an apparent toughness, K_{app}, the relation between the residual strength and crack size of a center-cracked panel is given by $\sigma_c = K_{app}\sqrt{\pi a}$. This residual strength is plotted as a function of total crack size as a Figure 4.44.

For small crack sizes, σ_c tends to infinity, but the residual strength at $a = 0$ cannot be larger than the material's ultimate tensile strength. When the material is sharply notched by a crack, the net section stress cannot become much higher than yield, especially when the material has a low work hardening rate. In other words, a small size panel (panel width, W, in Figure 4.44) will fail at net section yield. Although the fracture stress would be much higher according to its K_{app}.

With increasing panel size, the net section strength line will intersect the K_{app} line. For that case, Feddersen proposed[27] to use two linear tangents to the K curve. One tangent is drawn from $\sigma = \sigma_{ys}$, a = 0, the other from (σ = 0, 2a = W). A tangent to any point at the curve given by

$$\frac{d\sigma}{d(2a)} = \frac{d}{d(2a)} \left(\frac{K}{\sqrt{\pi a}}\right) = -\frac{\sigma}{4a} \quad . \qquad (4\text{-}64)$$

As shown in Figure 4.44 for the tangent through (σ_{ys}, 0), this yields

$$-\frac{\sigma_1}{4a_1} = \frac{\sigma_{ys} - \sigma_1}{4a_1} \quad \text{or } \sigma_1 = \frac{2}{3}\sigma_{ys}. \qquad (4\text{-}65)$$

Equation (4-65) shows that the left-hand tangency point is always at $\frac{2}{3}$ σ_{ys}, independent of K. The tangent through (0, W) (Figure 4.44) is defined by

$$-\frac{\sigma_2}{4a_2} = -\frac{\sigma_2}{W - 2a_2} \text{ or } 2a_2 = W/3 \quad . \qquad (4\text{-}66)$$

This means that the right-hand point of tangency is always at W/3. The right-hand tangent takes care of the finite size effect. Hence, K can simply be taken as $K = \sigma\sqrt{\pi a}$.

Consequently, the complete residual strength diagram can be constructed for any panel size if K_{app} is known. Two points can be taken at the curve; one at $\sigma = \frac{2}{3}\sigma_{ys}$, the other at W/3 and the tangents can be drawn to $(\sigma_{ys}, 0)$ and (0, W/3), respectively. The two points of tangency coincide when $\sigma_c = \frac{2}{3}\sigma_{ys}$ for 2a = W/3; i.e.,

$$\frac{2}{3}\sigma_{ys}\sqrt{\pi W/6} = K_{app} \quad \text{or} \quad W = \frac{27}{2\pi}\left(\frac{K_{app}}{\sigma_{ys}}\right)^2 \tag{4-67}$$

HENCE, PANELS SMALLER THAN THIS WILL FAIL BY NET SECTION YIELD. THEIR FAILURE POINT WILL BE BELOW THE K_{app} CURVE, WHICH MEANS THAT THEY CANNOT BE USED TO MEASURE K_{app}. OBVIOUSLY, THE SCREENING CRITERIA FOR A VALID TEST WOULD BE THAT THE FAILURE STRESS $\sigma_c < \frac{2}{3}\sigma_{ys}$ AND THE CRACK SIZE 2a < W/3. IF A TEST PANEL FAILS AT $\sigma_c \geq \frac{2}{3}\sigma_{ys}$ WHILE THE CRACK SIZE 2a = W/3, THE PANEL IS TOO SMALL TO DETERMINE K_{app}. SIMILAR ARGUMENTS CAN BE USED FOR K_i AND K_c. Note that if a K_{app} were calculated for such a case, its value would be smaller than the true K_{app} because of the lower failure stress.

Figure 4.44 shows that Feddersen's approach gives a fair representation of the data. The method is versatile in that it allows a simple characterization of plane stress and transitional residual strength. Presentation of K_{app} and/or K_c is sufficient to determine the residual strength for any crack size and panel size. Also, the method is based on stress intensity which makes it more universal. Finally, it gives a reasonable solution to the case of very small cracks where the fracture stress approaches yield. The left-hand tangent can be used for plane strain as well.

The plane stress fracture toughness (K_{app} and K_c) is usually determined from tests on center-cracked panels. The length of the (fatigue) crack should be smaller than W/3, and the failure stress lower than 2/3 σ_{ys} for a valid test. First, an unconservative estimate should be made of K_c. Then a specimen size can be selected to give $W > 27/2\pi(K_c/\sigma_{ys})^2$. This provides a better chance that $\sigma_c < 2/3\ \sigma_{ys}$ for $2a < W/3$, i.e., a valid test. The residual strength diagram can be constructed by using Feddersen's method as discussed.

In center-cracked thin plates, crack buckling may occur as a result of the compressive stress acting along the crack faces. This causes a reduction in fracture stress. IF SUCH BUCKLING WOULD BE CONSTRAINED IN THE ACTUAL STRUCTURE, ANTIBUCKLING GUIDES CAN BE APPLIED IN THE TEST IN ORDER TO ESTABLISH A RELEVANT FRACTURE TOUGHNESS VALUE. HOWEVER, IF BUCKLING WOULD NOT BE RESTRAINED IN SERVICE, ANTIBUCKLING GUIDES SHOULD NOT BE APPLIED IN THE TEST. As a rule of thumb, the uniform tensile stress, σ, at which crack-edge buckling will take place, is given by[11,28-30]

$$\sigma = \frac{\pi^2}{12}\frac{EB^2}{a^2} \qquad (4\text{-}14)$$

where E is Young's modulus, B is plate thickness, and 2a is the size of the central crack.

The plane stress fracture toughness data in the Damage Tolerance Design Handbook[20] are categorized according to whether or not buckling restraints were applied.

4.3.1.1.5 Plasticity Effects

Elastic solutions of crack-tip stress fields show a stress singularity at the crack tip, which implies that the stresses will always be infinite (as shown by Equation 4-1). Since structural materials deform plastically above the yield stress, a plastic zone will develop at the crack tip. As a consequence, the crack-tip stress will be finite.

A rough estimate of the magnitude of the plastic zone easily can be made. The elastic stress in the Y direction along Y = 0 is given as[11]

$$\sigma_y' = \frac{K}{\sqrt{2\pi r}} \text{ ; for a small center crack in a wide plate, } \sigma_y = \frac{\sigma\sqrt{\pi a}}{\sqrt{2\pi r}} . \tag{4-68}$$

This stress distribution is shown diagrammatically in Figure 4.45a. It is assumed that nowhere can the stress be higher than the yield stress, σ_{ys} (Figure 4.45b). The distance from the crack tip, r_p, to which the elastic stresses are above yield is found by substituting $\sigma_y = \sigma_{ys}$ in Equation (4-68),

$$\sigma_{ys} = \frac{\sigma\sqrt{\pi a}}{\sqrt{2\pi r_p}} \text{ or } r_p = \frac{\sigma^2 a}{2\sigma_{ys}^2} = \frac{K^2}{2\pi\sigma_{ys}^2} \tag{4-69}$$

The crack-tip plasticity gives rise to slightly larger displacement than in the elastic case. This is sometimes accounted for by using an apparent crack size, $a^* = a + r_p$. The stress intensity then becomes

$$K_I = \beta\sigma\sqrt{\pi a^*} = \beta\sigma\sqrt{\pi(a + r_p)} = \beta\sigma\sqrt{\pi a + \sigma K_I^2/\sigma_{ys}^2} \ . \qquad (4\text{-}70)$$

When the applied stress is half the yield stress, the plastic zone size, r_p, is 0.125a. AS LONG AS THE PLASTIC ZONE IS SMALL COMPARED WITH THE CRACK SIZE, THE STRESS DISTRIBUTION WILL BE AFFECTED ONLY SLIGHTLY BY THE PLASTIC ZONE. PARTICULARLY, THE STRESS DISTRIBUTION OUTSIDE THE PLASTIC ZONE IS STILL GOVERNED BY K. SINCE THE SAME K ALWAYS GIVES RISE TO THE SAME PLASTIC ZONE SIZE [EQUATION (4-69)], THE STRESSES AND STRAINS INSIDE THE PLASTIC ZONE WILL BE A DIRECT FUNCTION OF THE STRESS-INTENSITY FACTOR. HENCE, K STILL CAN BE USED AS THE GOVERNING PARAMETER FOR CRACK GROWTH AND FRACTURE.

Several more rigorous solutions for the plastic zone[12-15] indicate that the actual plastic zone shape is somewhat different from the idealized circular zone in Figure 4.45b. Experimental verification is difficult, because elastic and plastic strain cannot easily be distinguished. FROM THE POINT OF VIEW OF DAMAGE TOLERANCE ANALYSIS, THE ROUGH ESTIMATE OF PLASTIC ZONES AS IN EQUATION (4-69) IS ADEQUATE.

The state of stress affects the plastic zone size, plane strain being associated with a smaller plastic zone than plane stress. At the same time, the size of the plastic zone largely affects the state of stress. The material in the plastic zone wants to contract in the thickness direction (more than in the elastic case, because of the condition of constant volume during plastic flow). When the plastic zone is large compared with the plate thickness, yielding in the thickness direction can take place freely. This promotes plane stress. If the plastic zone is small compared with the thickness, yielding in the thickness direction will be constrained. As a result, a small plastic zone is under plane strain. Plane stress can develop when the plastic zone is of the order of the plate thickness.

The surface of a plate always will be in plane stress because a stress perpendicular to the free surface cannot exist. If the plate is very thick, the plane strain region in the interior will be large with respect to the plane-stress surface regions. Thus, plane strain behavior will dominate. As a general rule, this is the case when the (plane strain) plastic zone is only about 2 percent of the plate thickness. The plastic zone size can be expressed in terms of K. For plane strain, it is approximately $K^2/6\pi\sigma_{ys}^2$. Hence, the plane strain condition is that $B \geq 2.5K^2/\sigma_{ys}^2$, where B is the plate thickness.

Increase of the stress increases the plastic zone size. Full plane stress can develop when the plastic zone size is on the order of the plate thickness (i.e., $B \simeq \alpha K^2/\sigma_{ys}^2$, where α is on the order of 0.1 - 0.2). Between these two thickness conditions, there will be a gradual transition from full plane strain to full plane stress.

THE ABOVE CRITERIA FOR THE STATE OF STRESS ARE NOT APPLICABLE TO CRACKS WITH A CURVED FRONT (I.E., CORNER CRACKS AND SURFACE FLAWS). THE

CURVATURE MAINTAINS A STRESS TANGENTIAL TO THE CRACK FRONT. AS A RESULT, THE GREATER PART OF THE CRACK FRONT IS ALWAYS IN PLANE STRAIN.

A thick plate under plane strain will fracture when the stress intensity reaches the critical value, K_{Ic}. When the plate thickness, B, exceeds $2.5\ K_{Ic}^2/\sigma_{ys}^2$, the critical value will be essentially independent of plate thickness, because the plane strain part is large compared with the plane stress region. The exception to this is when metallurgical factors produce a different micro, structure in very thick section materials.

4.3.1.1.6 Summary

In finding and applying a fracture toughness value for a given thickness, the following guidelines apply.

a. FOR RELIABLE RESIDUAL STRENGTH PREDICTIONS, IT IS A PRE-REQUISITE TO USE TOUGHNESS DATA RELEVANT TO THE HEAT AND THICKNESS THAT WILL ACTUALLY BE USED IN THE DESIGN.

b. Alternatively, K_{Ic} can be taken as a safe lower boundary.

c. Fracture toughness data for a variety of alloys and thicknesses can be found in the Damage Tolerance Design Handbook[20]. On the basis of these data, reasonable estimates of the toughness can be made for a given alloy and a given thickness for application in the early design stages.

d. If insufficient data are available for a given material, the linear model proposed by Anderson[16] might be used to obtain a rough number for the toughness. This requires knowledge of K_{Ic} and of the toughness K_c for one other thickness. A linear interpolation between K_{Ic} [at a thickness $2.5\ (K_{Ic}/\sigma_{ys})^2$] and K_c (at the given thickness) yields K_c values for intermediate thicknesses.

4.3.1.2 Crack Growth Resistance Curves

Important to the development of any materials fracture criterion are those environments or material properties which affect the determination of a given fracture parameter. This is equally important in the development of any structural fracture criteria. In a given aircraft structure there can be two types of fracture criteria - so-called skin critical and stiffener critical cases. It is important to consider both criterion in any complete fracture or residual strength analysis. However, in the absence of any fatigue cracks in the stiffener the more important problem deals with obtaining the necessary data to assess if a skin or stiffener critical case governs. A detailed procedure for obtaining this data is given in Reference 102, Part II, Volume II.

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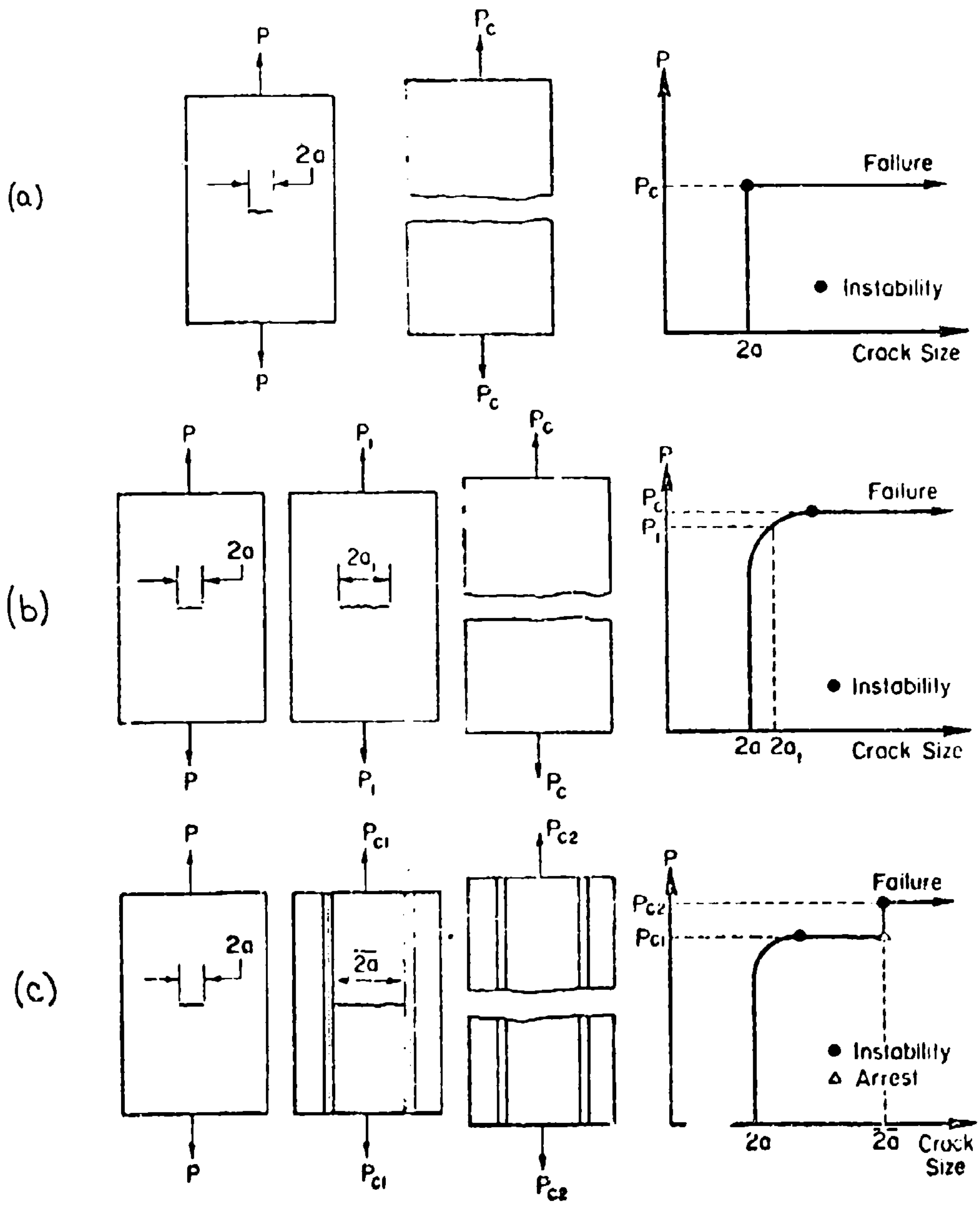


Figure 4.1 Possible Cases of Fracture

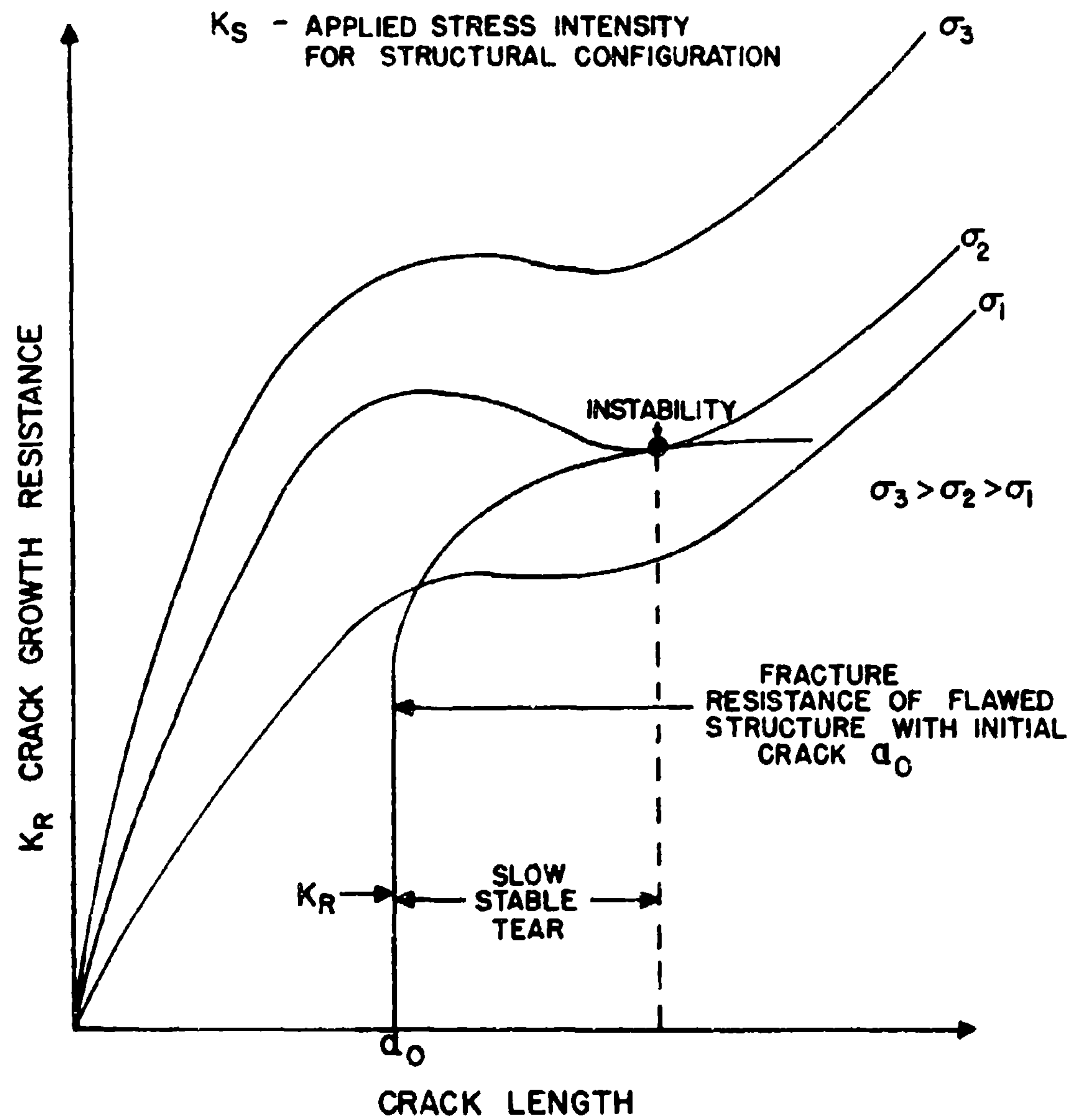


Figure 4.2 Crack-Growth Resistance as a Failure Criterion

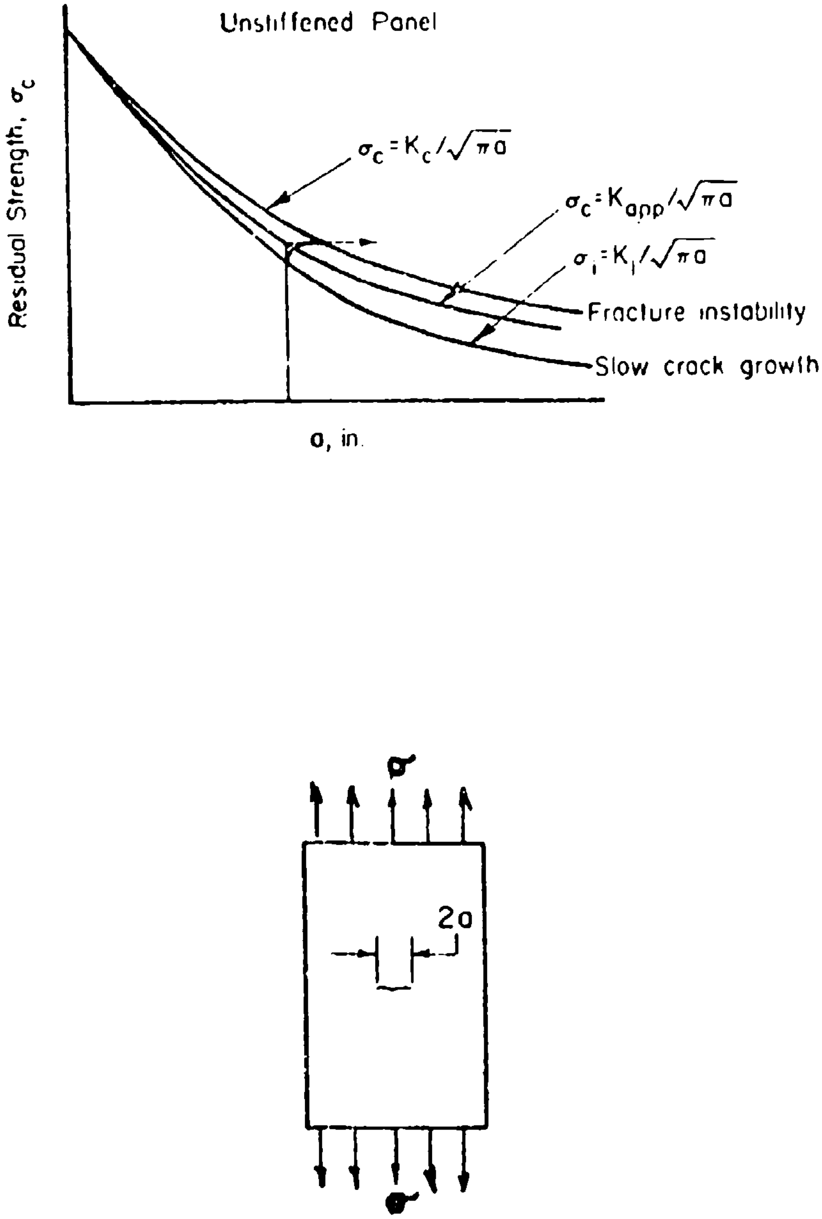
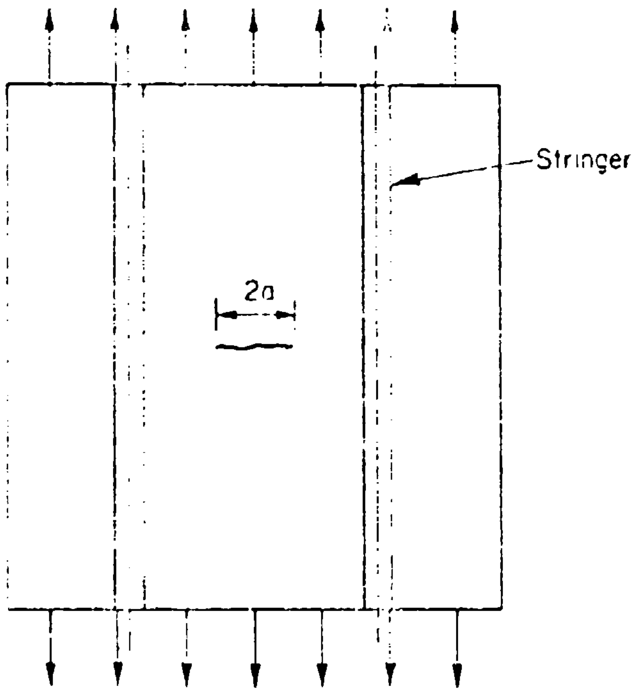


Figure 4.3 Residual-Strength Diagram for an Unstiffened Panel



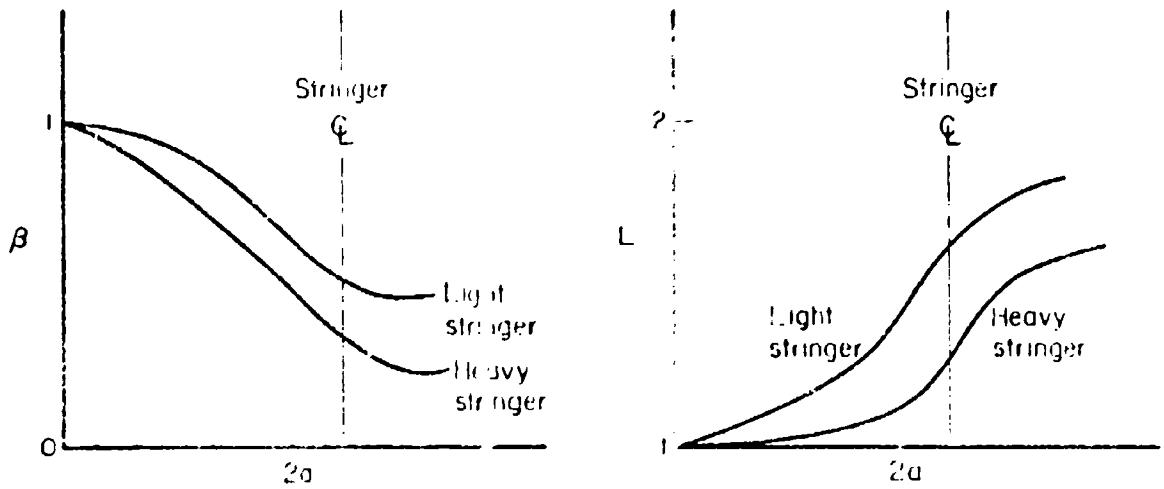


Figure 4.4 Variation of β and L with Crack Length in Stiffened Panel with Crack Between Stiffeners

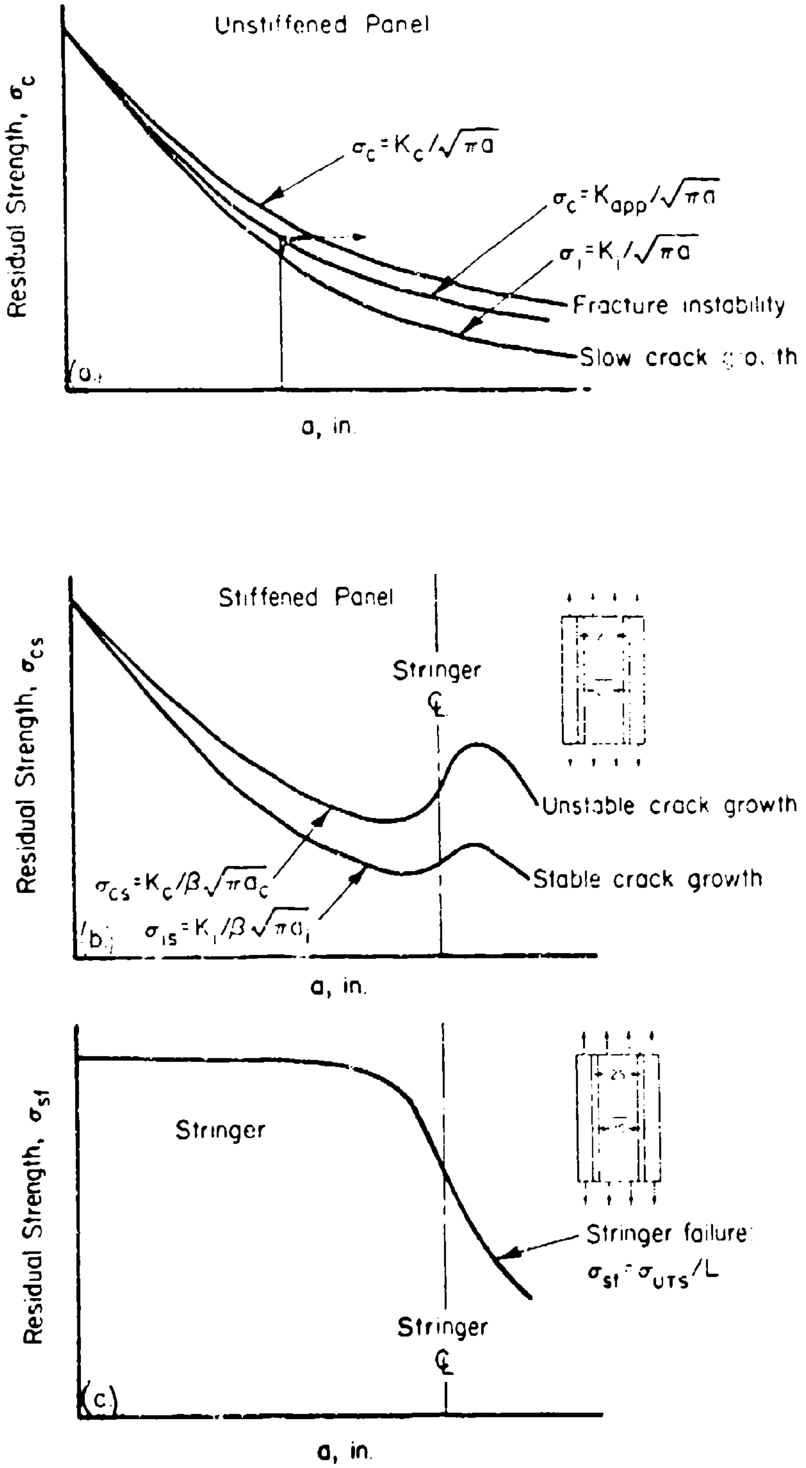


Figure 4.5 Elements of Residual-Strength Diagram

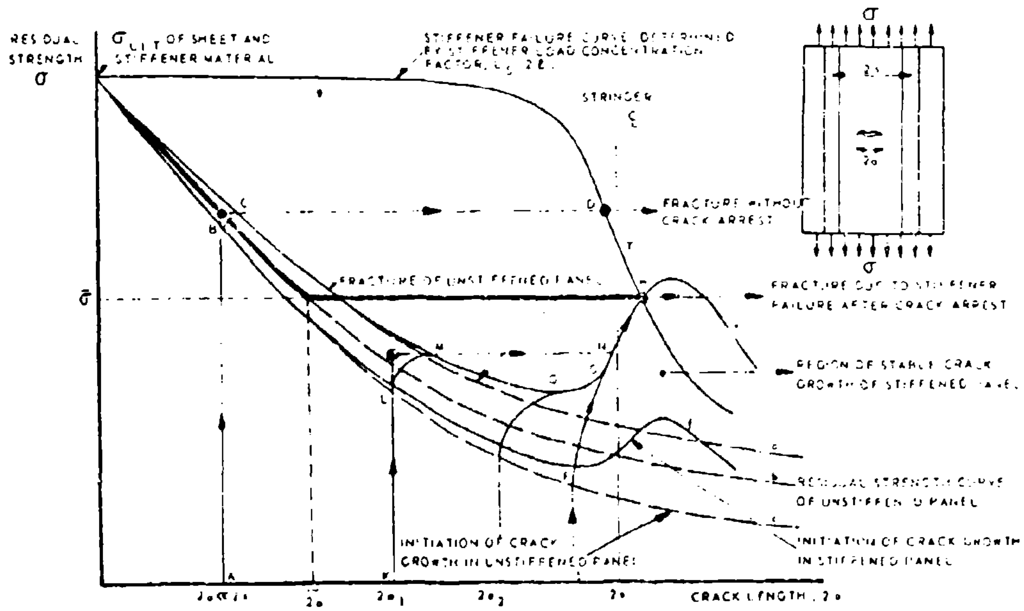


Figure 4.6 Residual-Strength Diagram for Simple Stiffened Panel

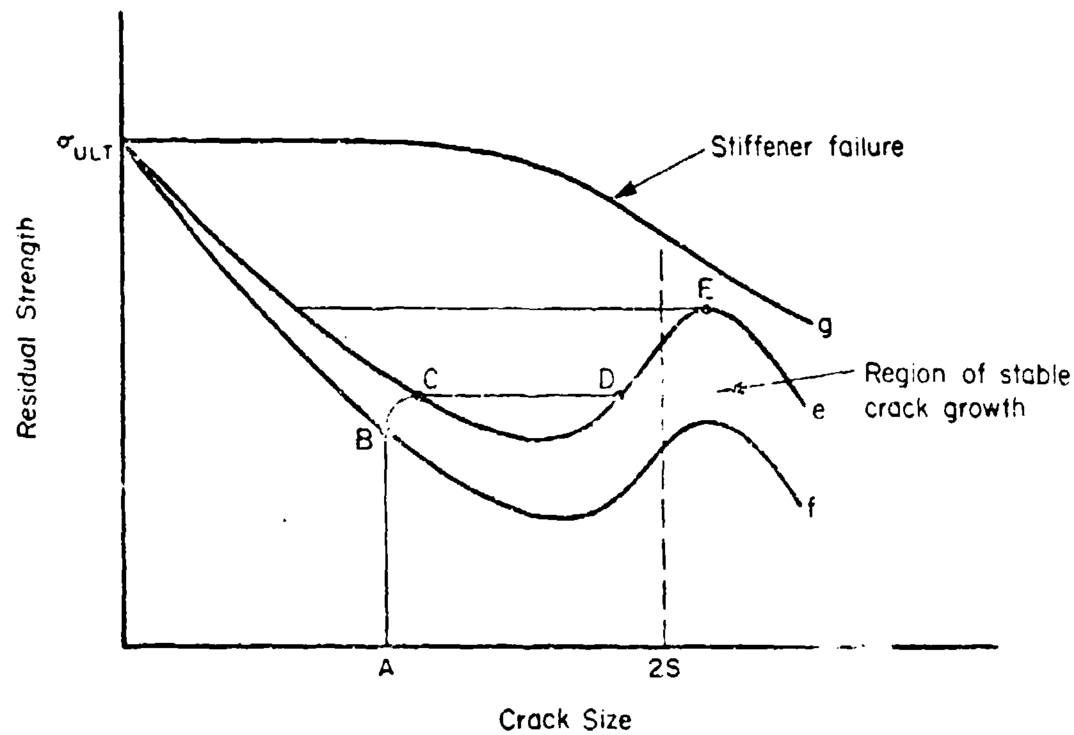


Figure 4.7 Panel Configuration with Heavy Stringers; Skin-Critical Case

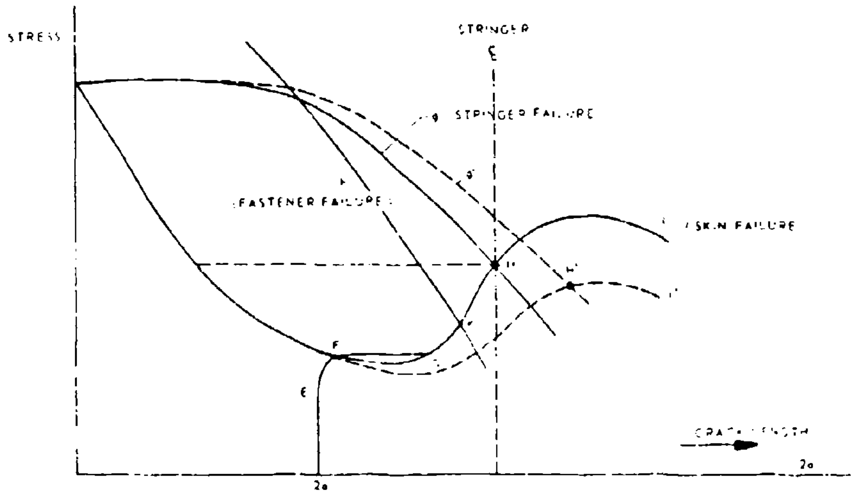


Figure 4.8 Criterion for Fastener Failure

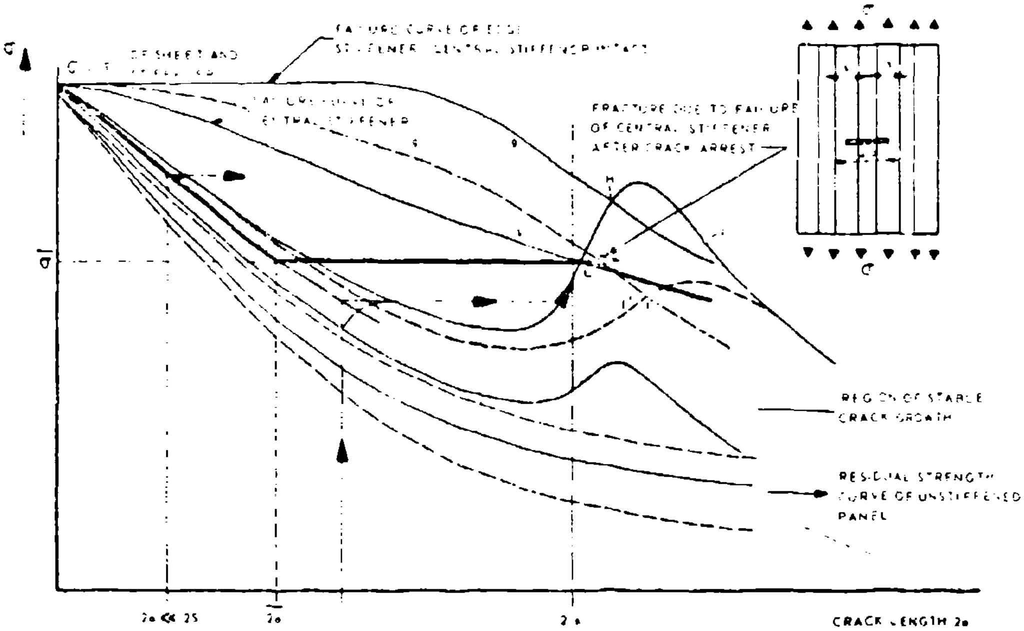


Figure 4.9 Residual-Strength Diagran for
a Panel with Three Stiffeners
and a Central Crack

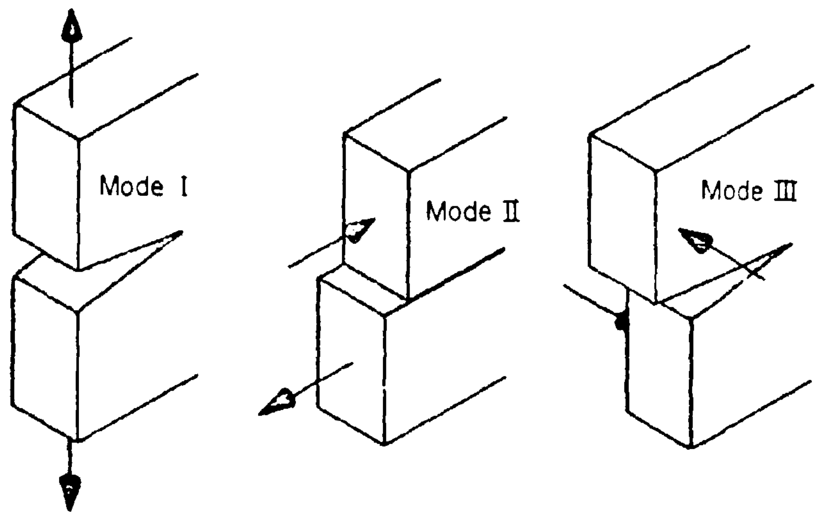
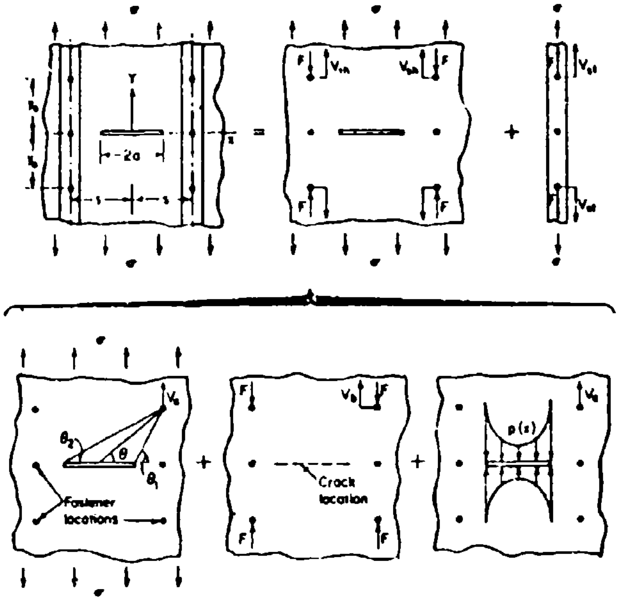
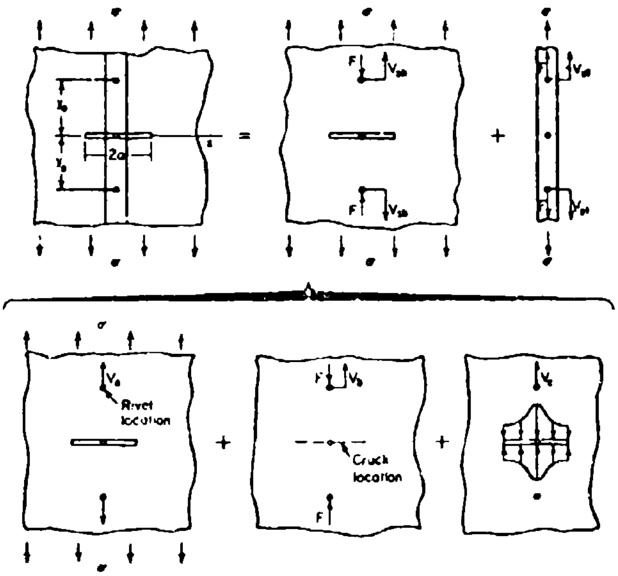


Figure 4.10 Three Modes of Cracking



a. Crack Between Stringers



b. Crack Acros· Stringer

Figure 4.11 Analysis of Stiffened Panel

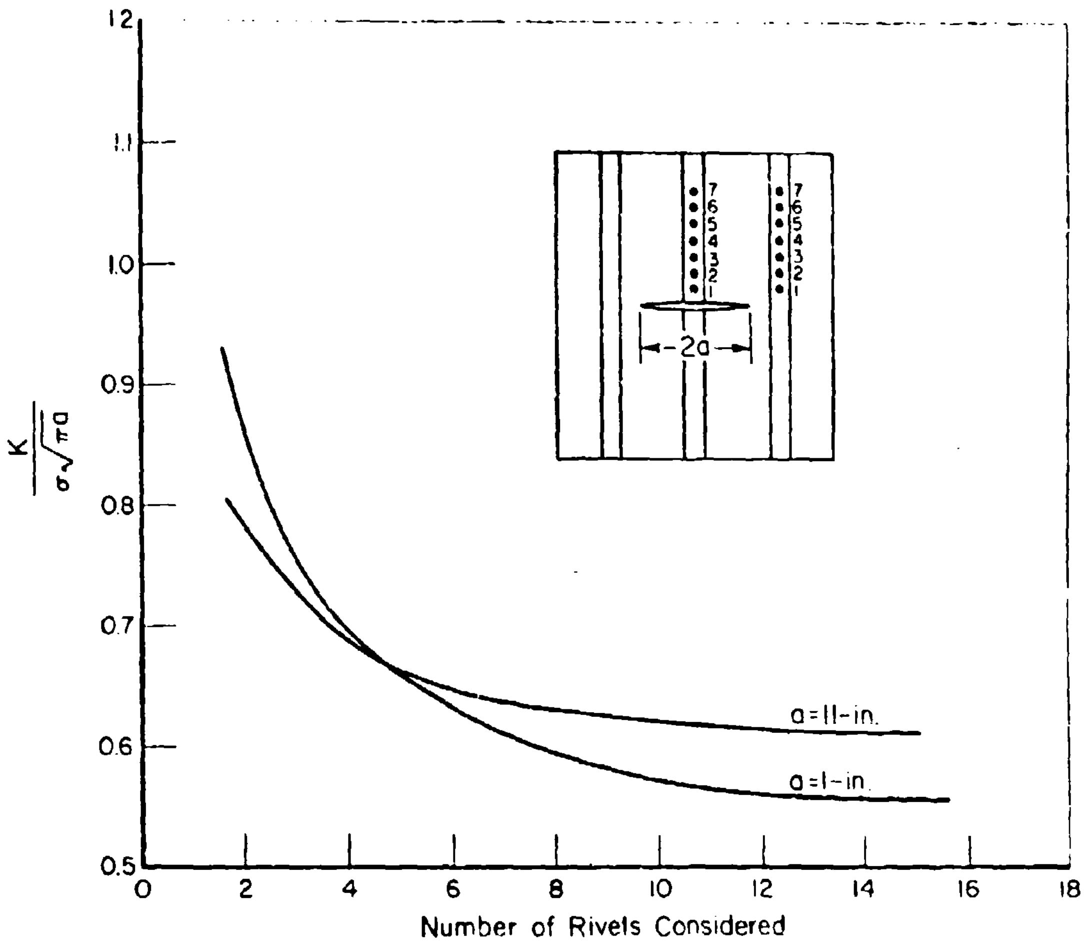


Figure 4.12 Effect of Number of Fasteners Included in Analysis on Calculated Stress Intensity

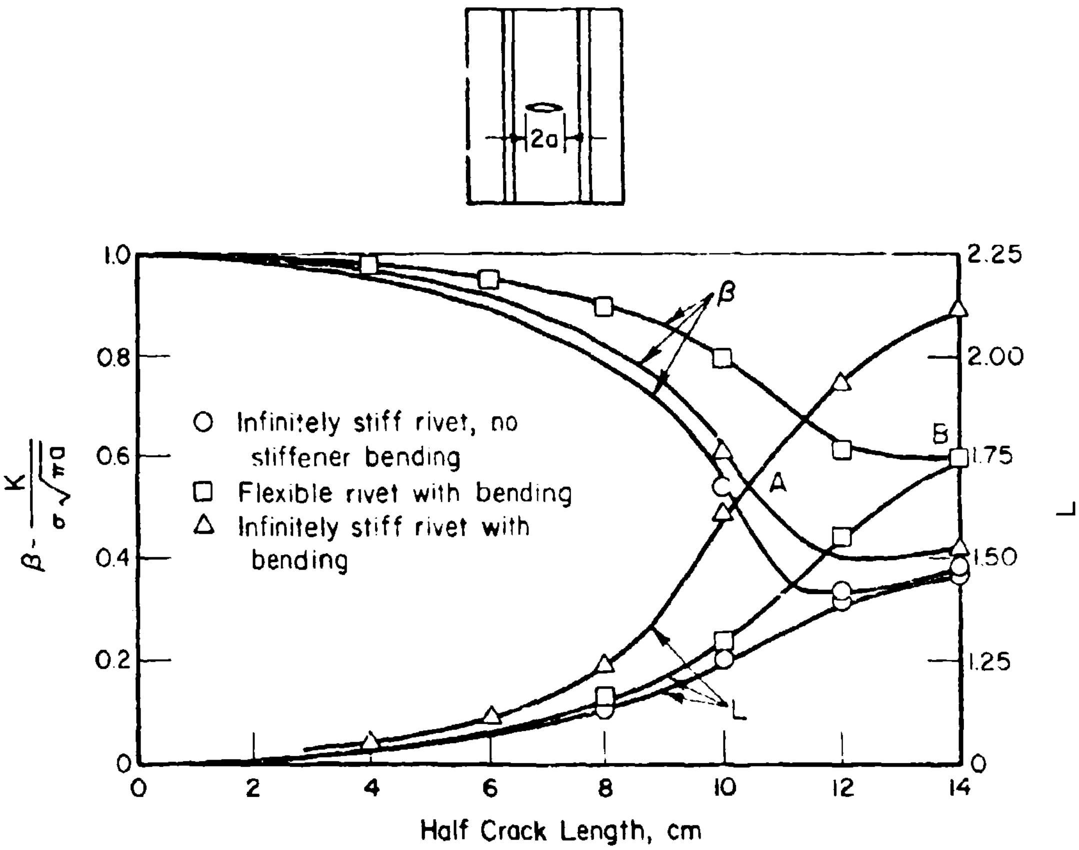
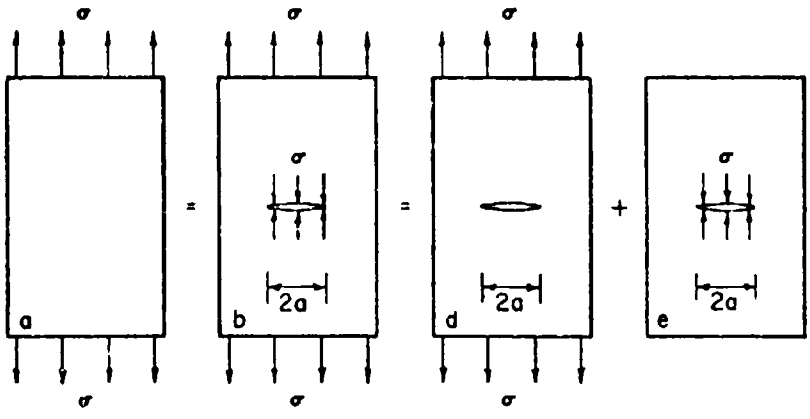
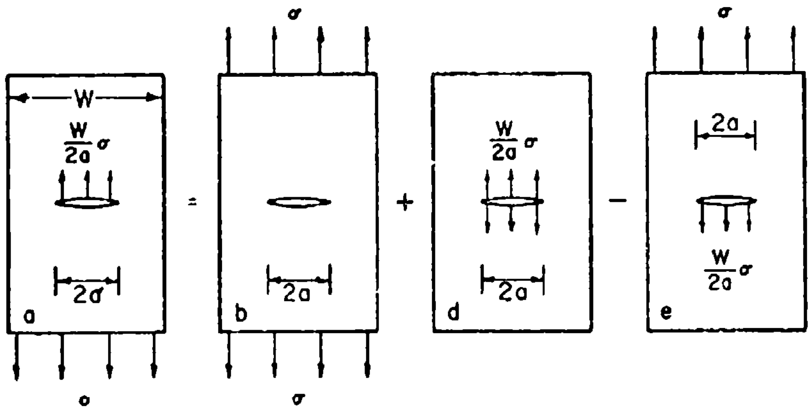


Figure 4.13 Skin-Stress-Reduction β and Stringer-Load-Concentration L as Affected by Fastener Flexibility and Stiffener Bending



$K_a = K_b = K_d + K_e = 0 \qquad K_e = -K_d$

Figure 4.14 Illustration of Superposition Principle



$K_a = K_b + K_d - K_e \qquad 2K_a = K_b + K_d \qquad K = \frac{K_b + K_d}{2}$

Figure 4.15 Application of Superposition Principle

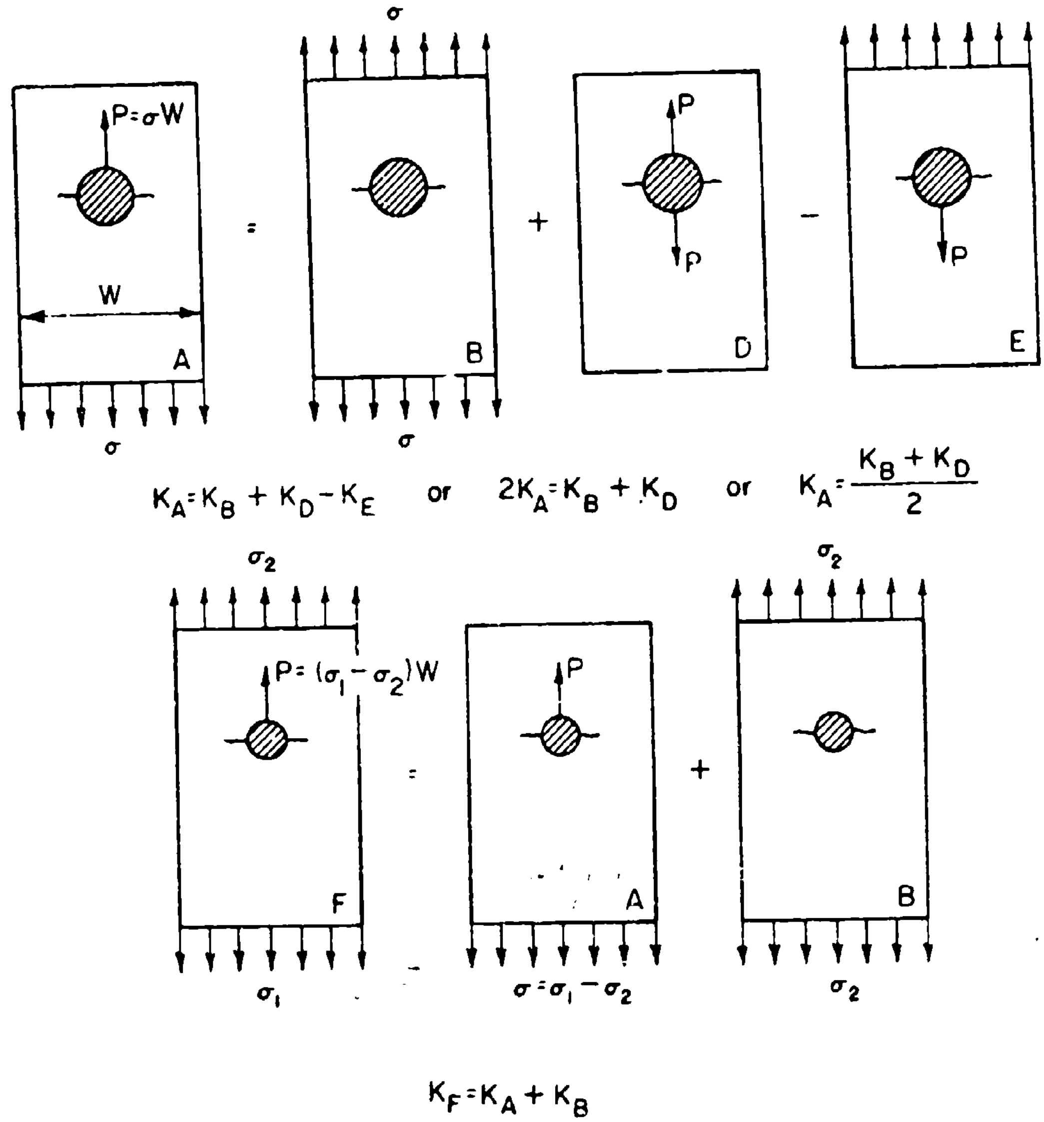


Figure 4.16 Stress-Intensity Factor for Pin-Loaded Hole Obtained by Superposition

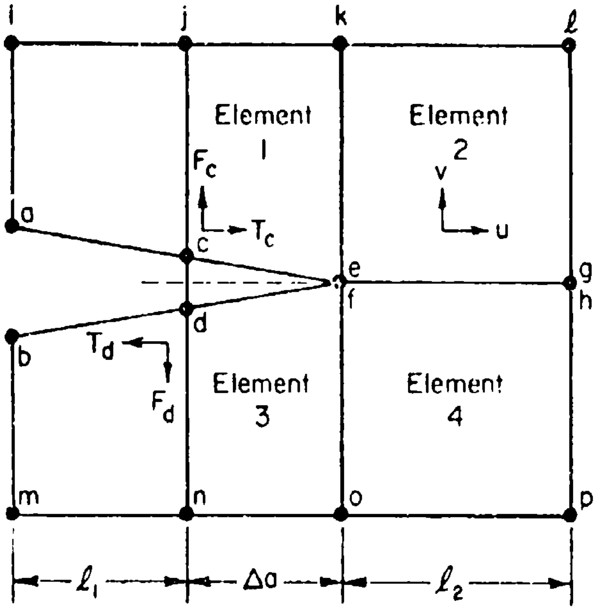


Figure 4.17 Finite-Element Nodes Near Crack Tip

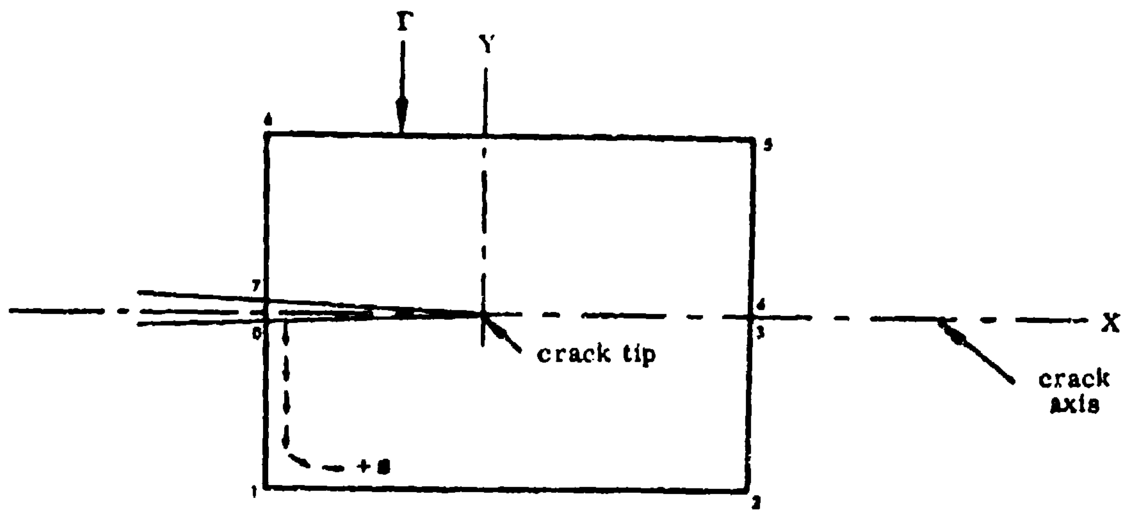


Figure 4.18 Rectangular Path for J Calculation

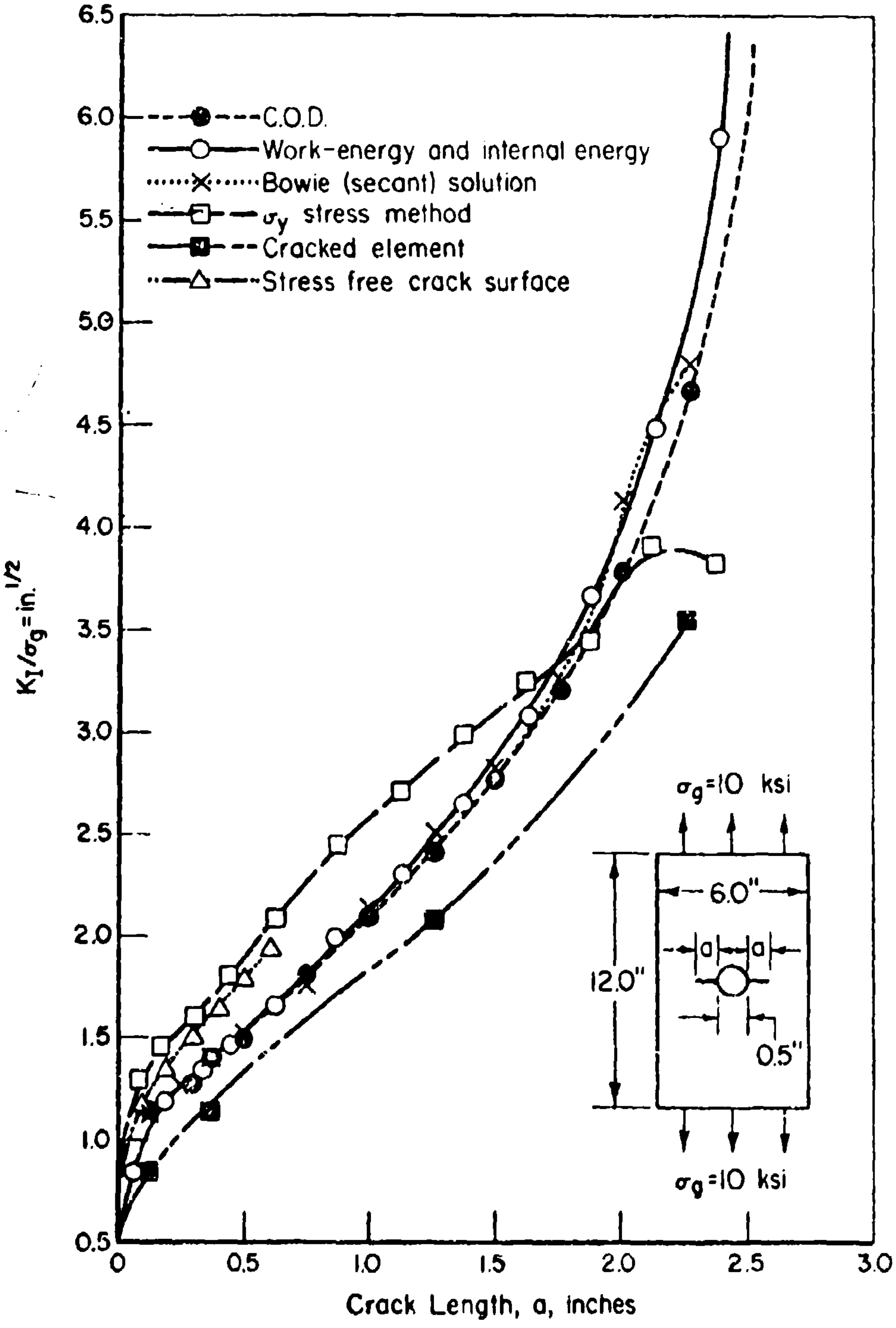


Figure 4.19 Stress-Intesity Factor Versus Crack Size (Coarse Grid Finite-Element Model)

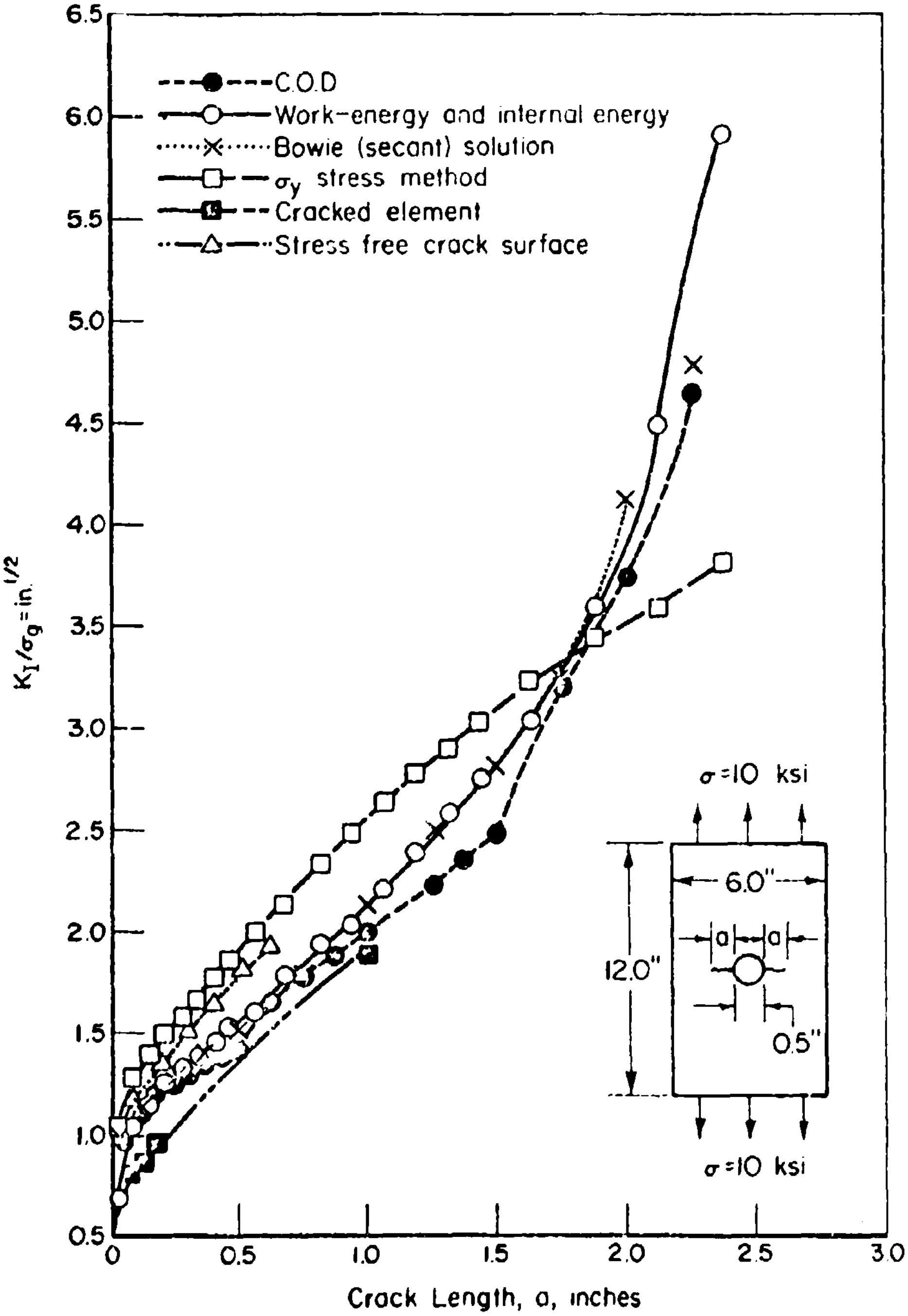


Figure 4.20 Stress-Intensity Factor Versus Crack Size (Fine Grid Finite-Element Model)

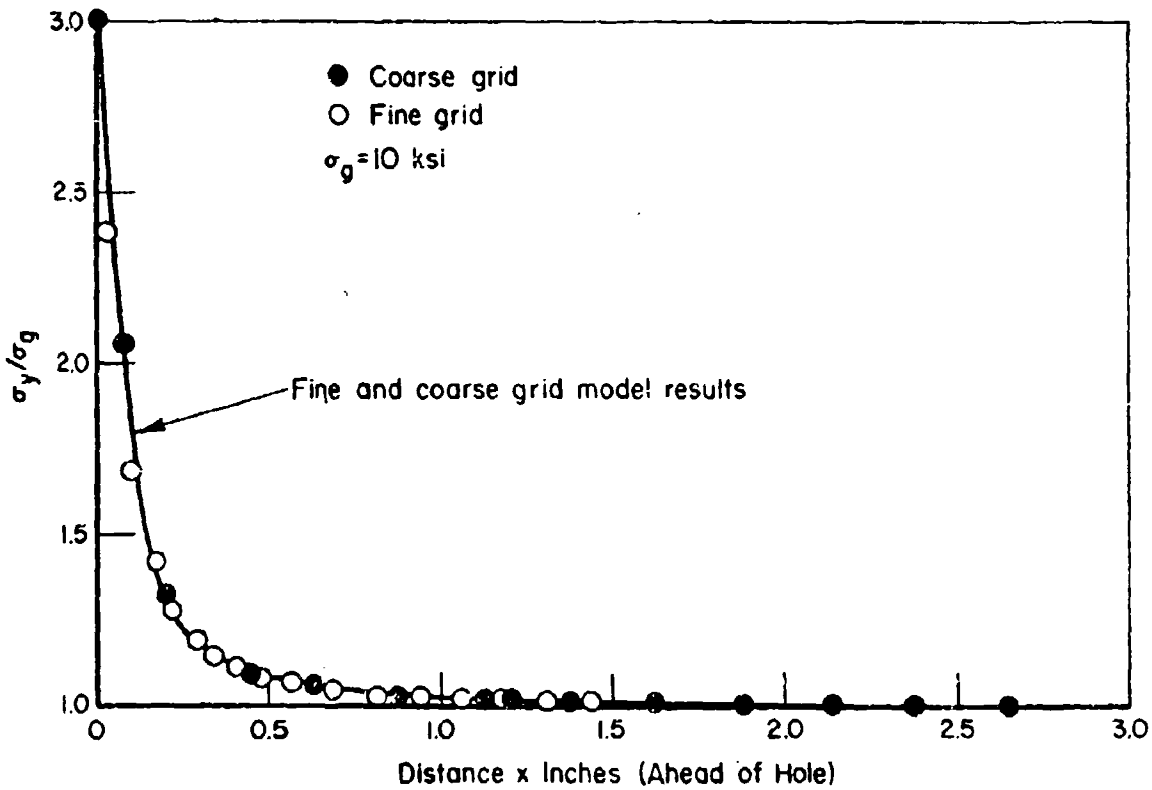


Figure 4.21 Variation of σ_y Stress From Edge of Hole

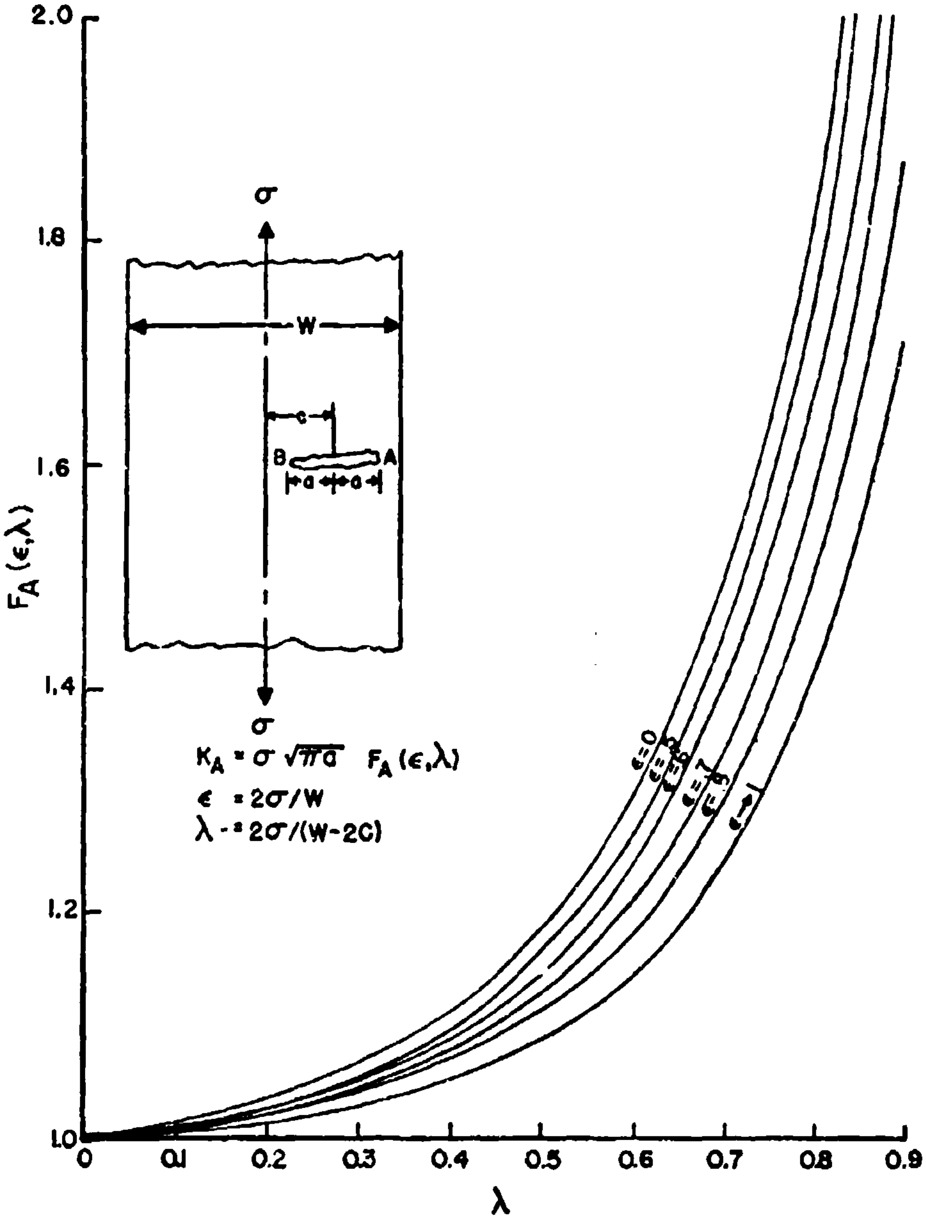


Figure 4.22 Finite Width Correction-Eccentric Crack, $F_a(\epsilon, \lambda)$

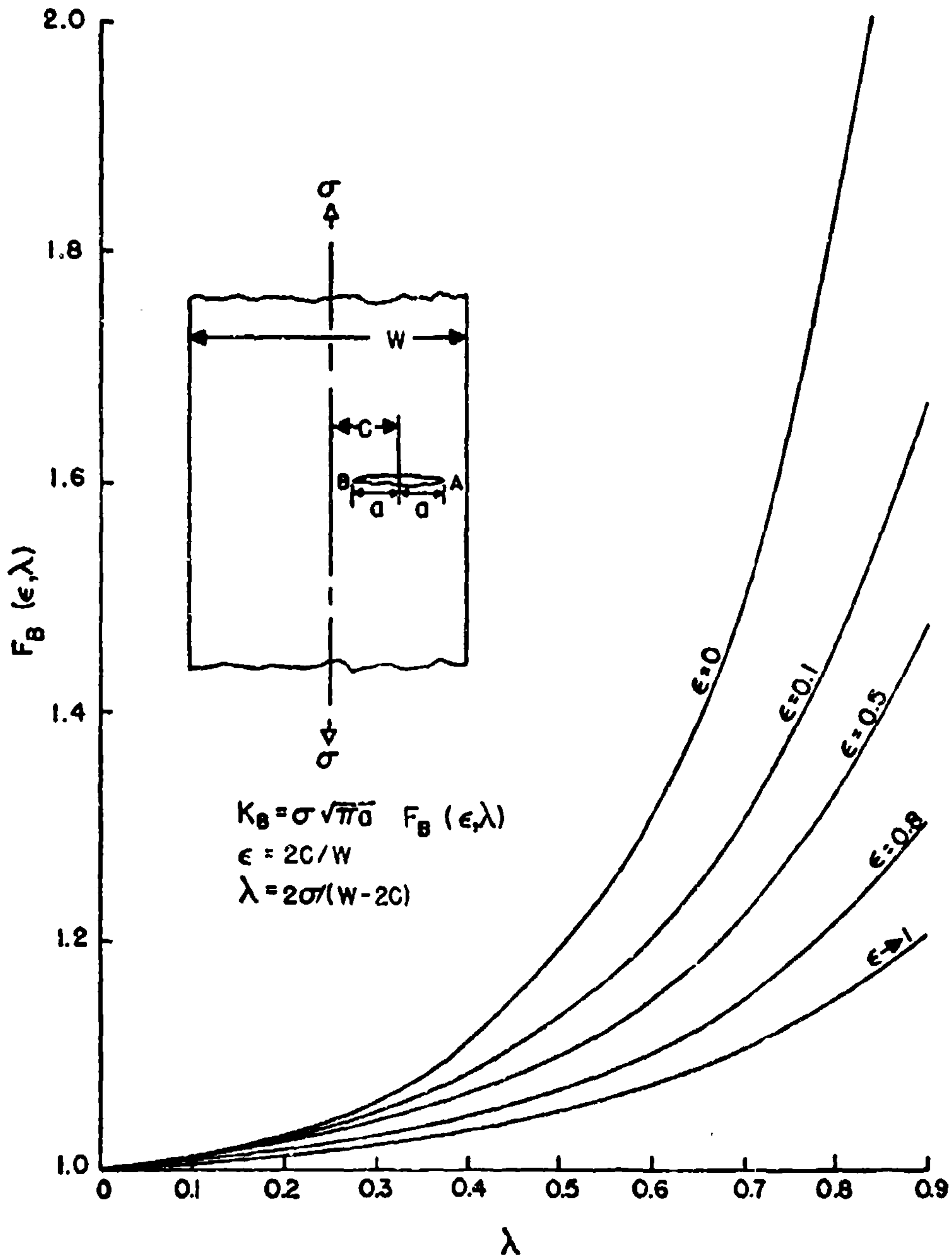


Figure 4.23 Finite Width Correction-Eccentric Crack, $F_B(\varepsilon, \lambda)$

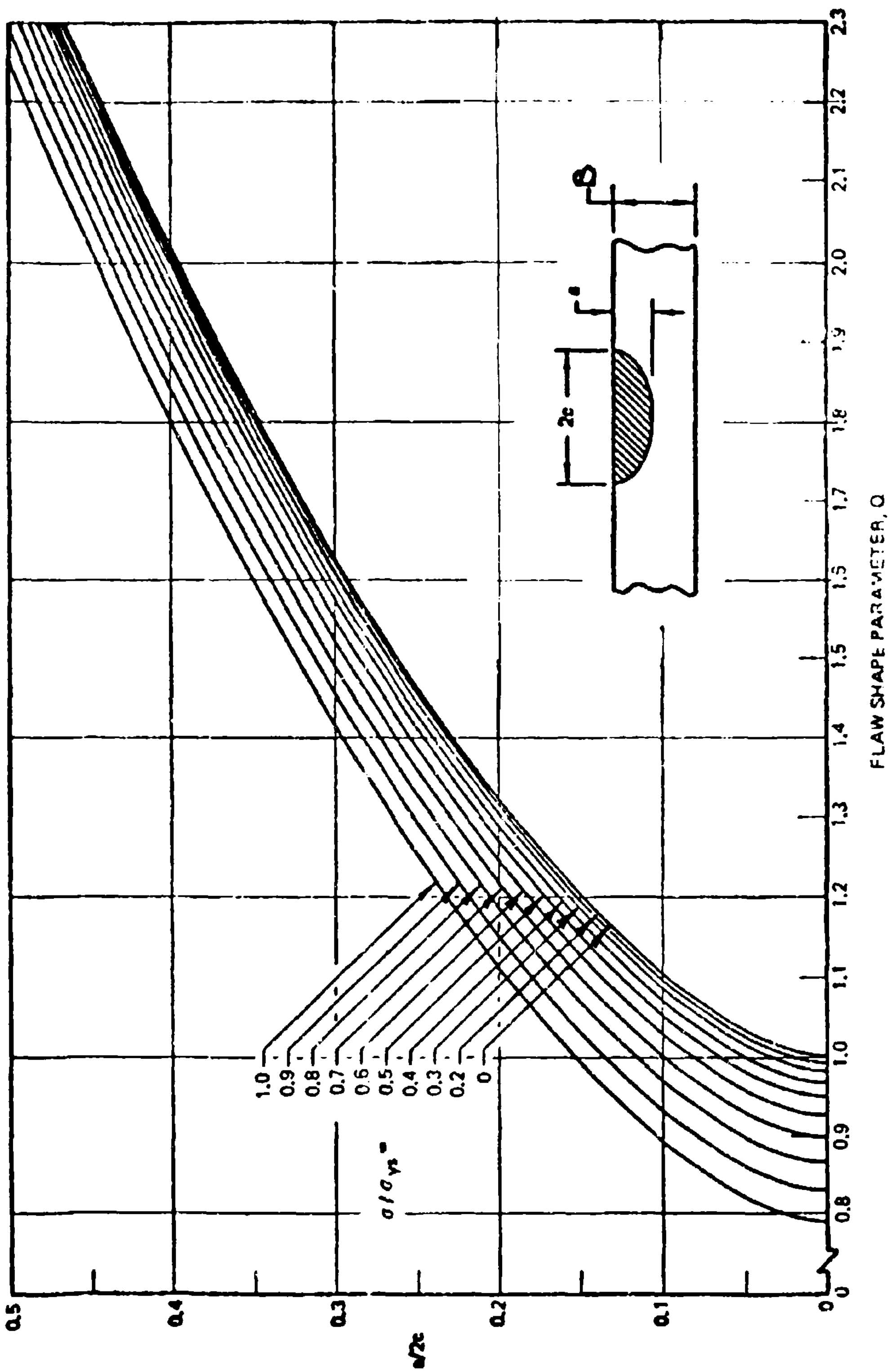


Figure 4.24 Shape Parameter Curves for Surface and Internal Flaws

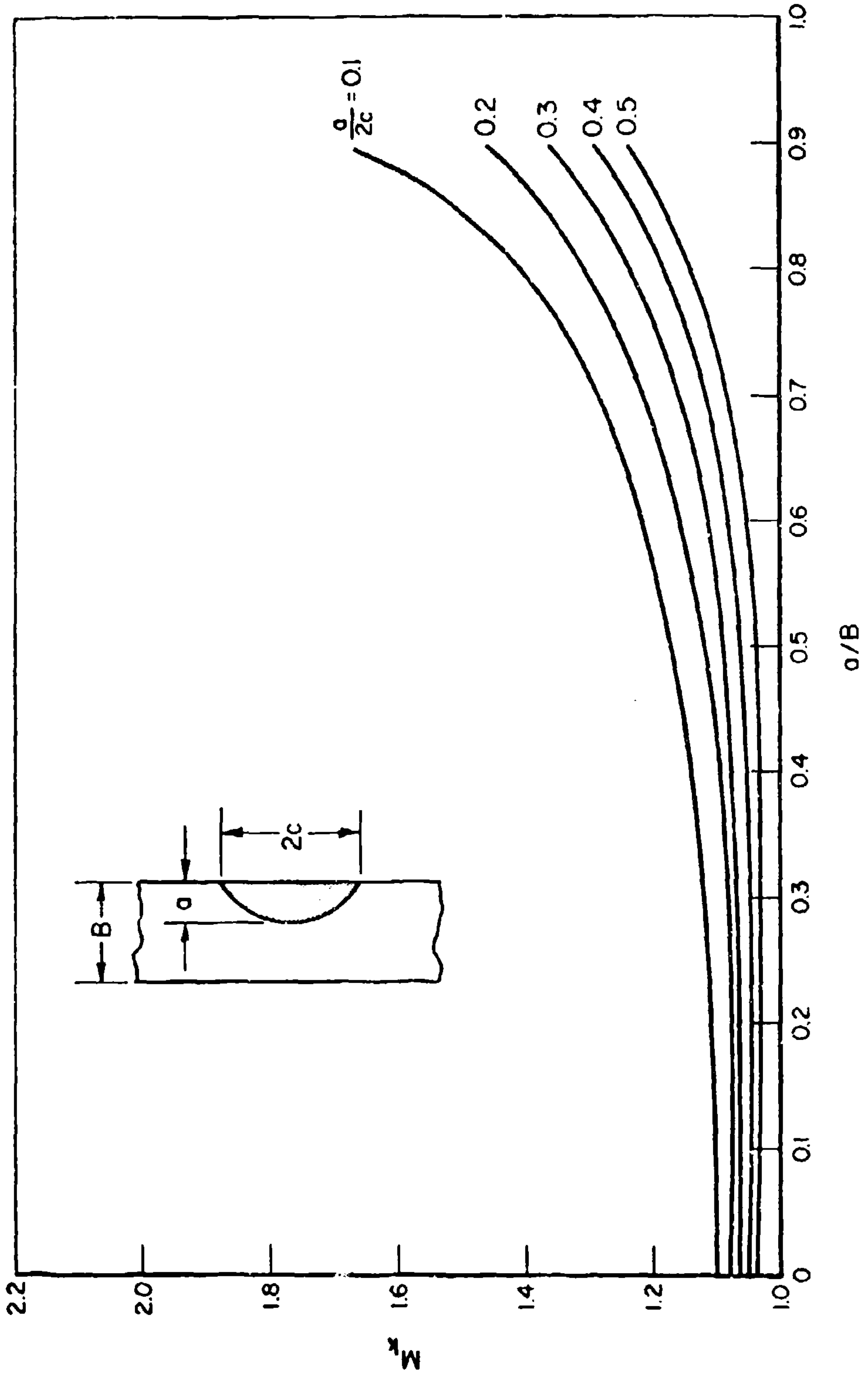


Figure 4.25 Back-Free Surface Correction

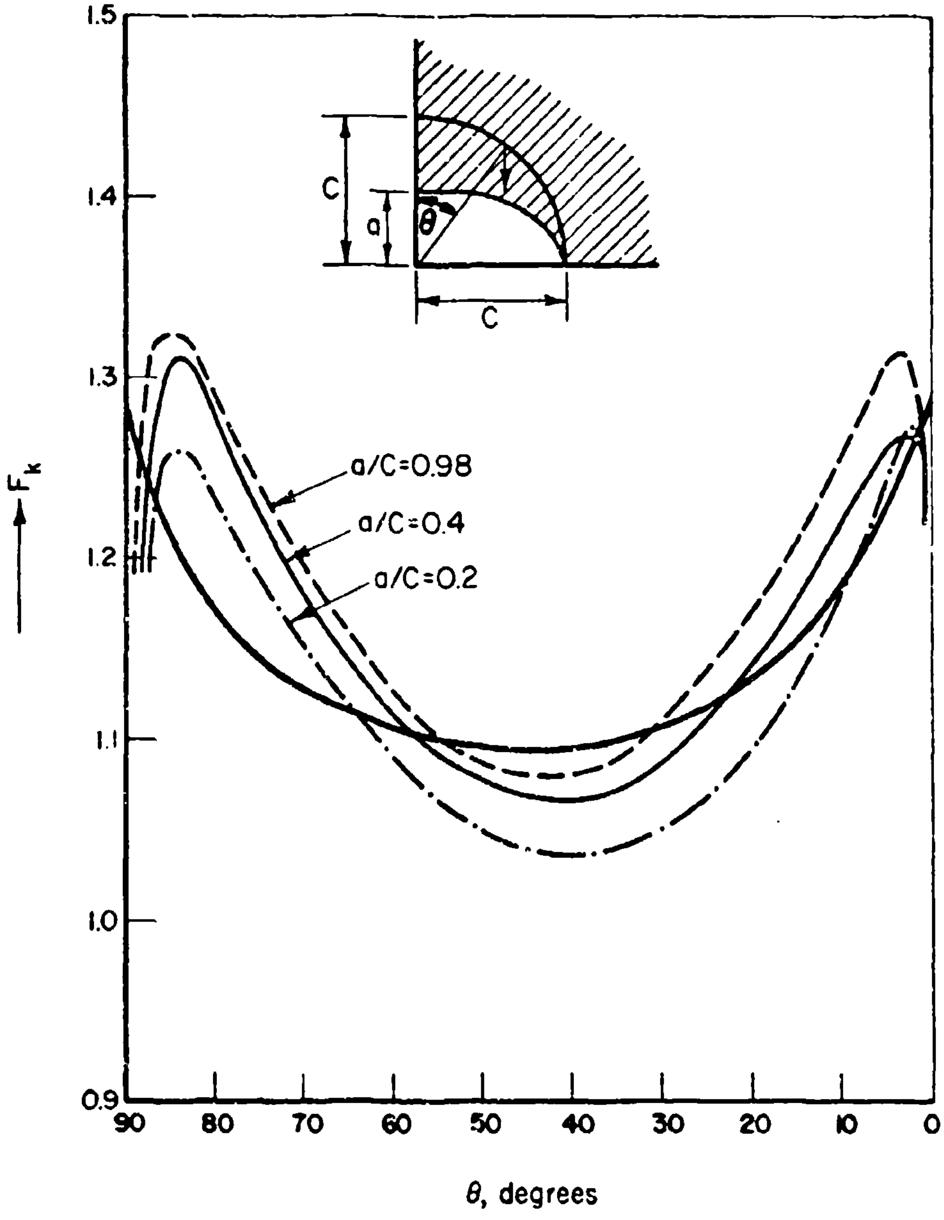


Figure 4.26 Stress-Intensity Correction for Corner Flaw

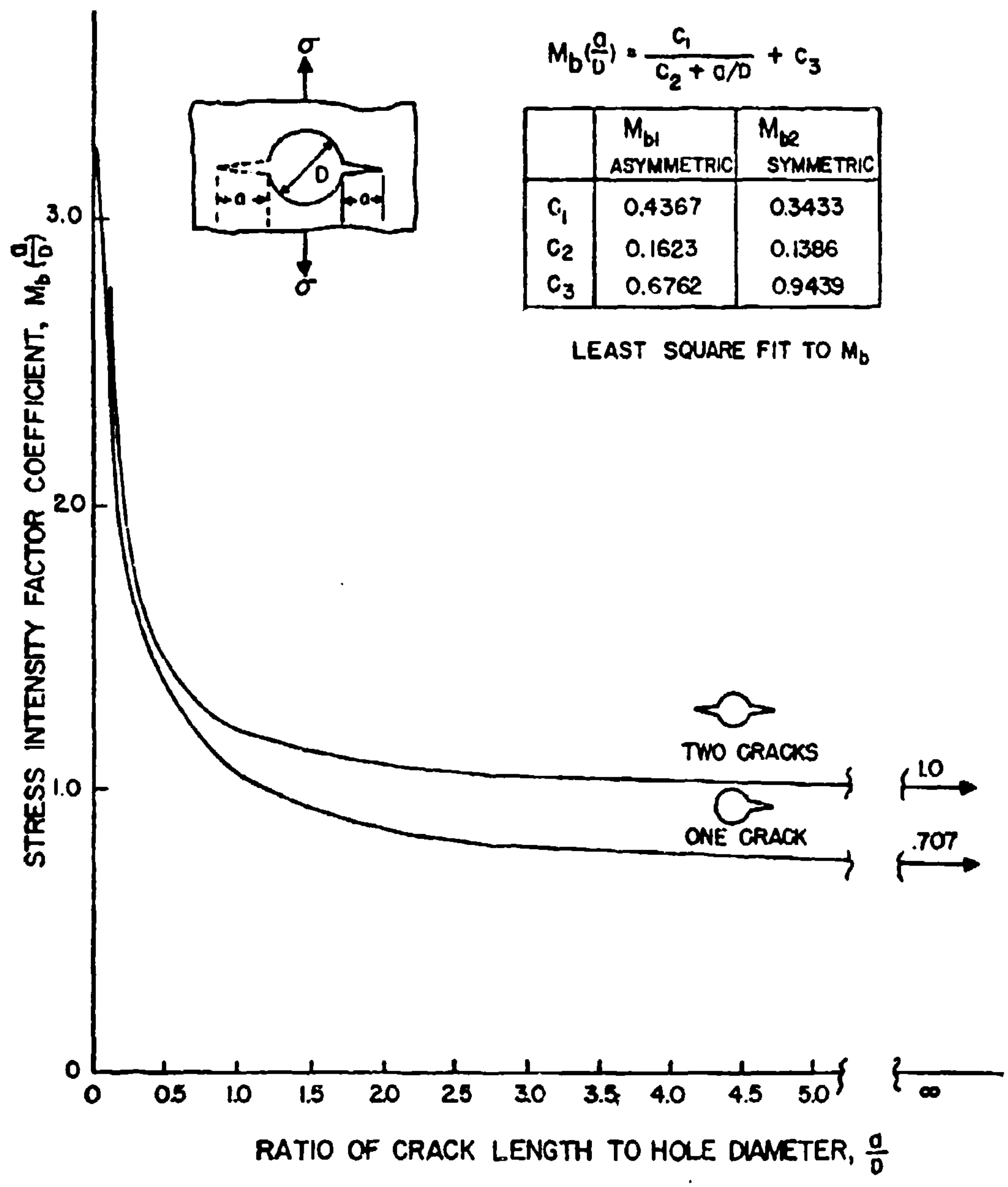


Figure 4.27 Bowie Solutions for Radial Through Cracks

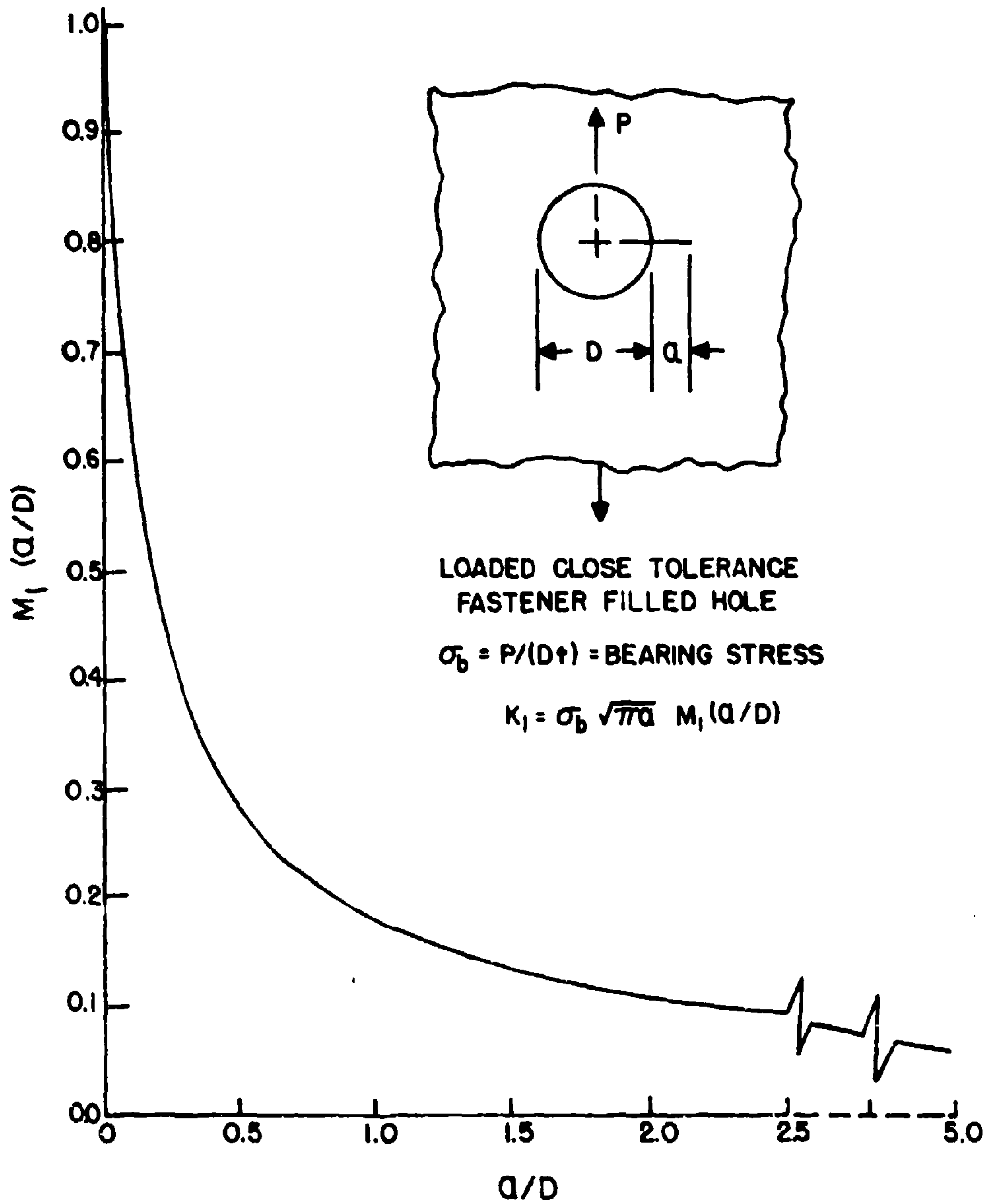


Figure 4.28 Nondimensionalized Stress Intensity Factors for One Through Crack Originating from a Loaded Close Tolerance Fastener

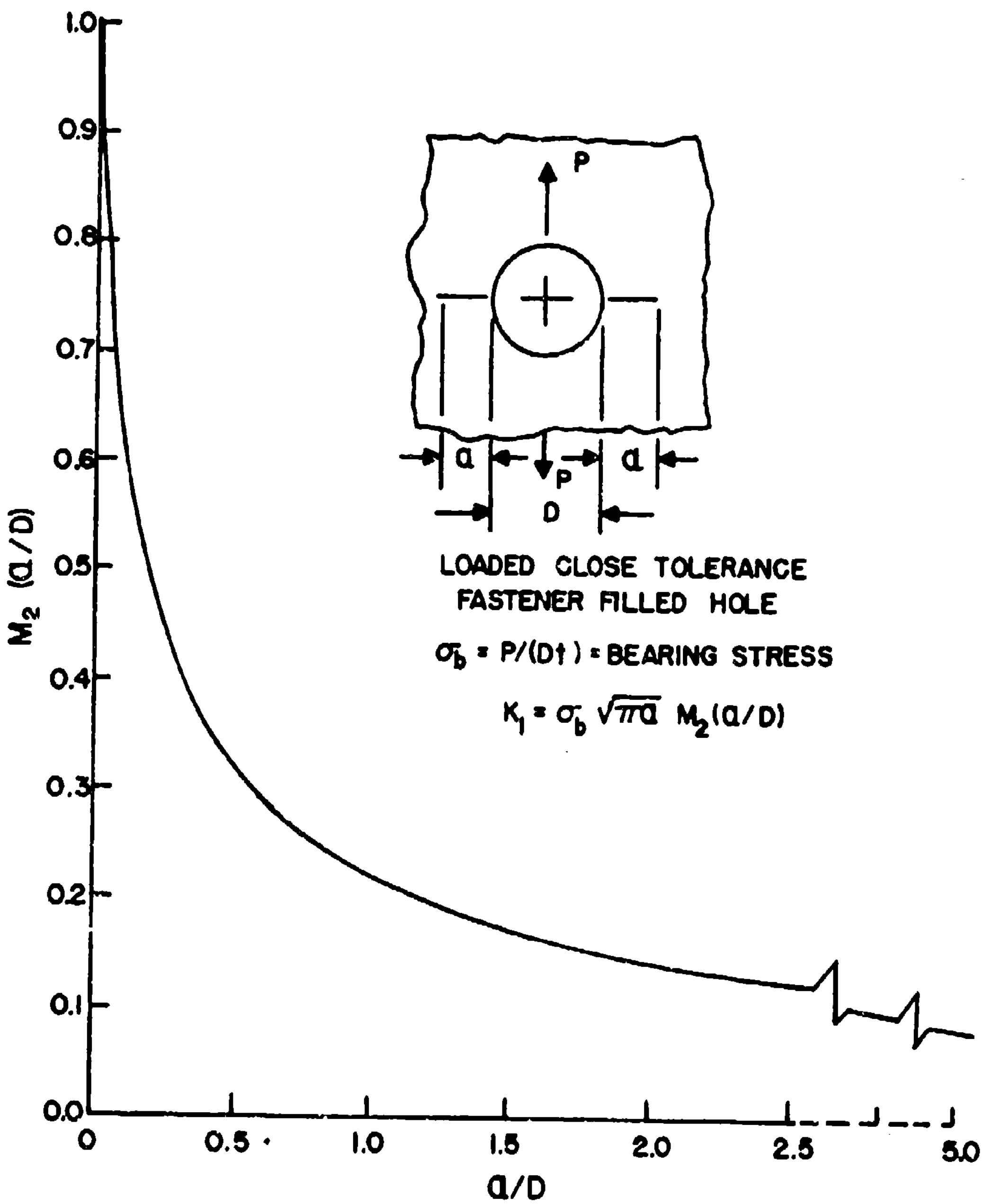


Figure 4.29 Nondimensionalized Stress Intensity Factors for Two Through Cracks Originating from a Loaded Close Tolerance Fastener

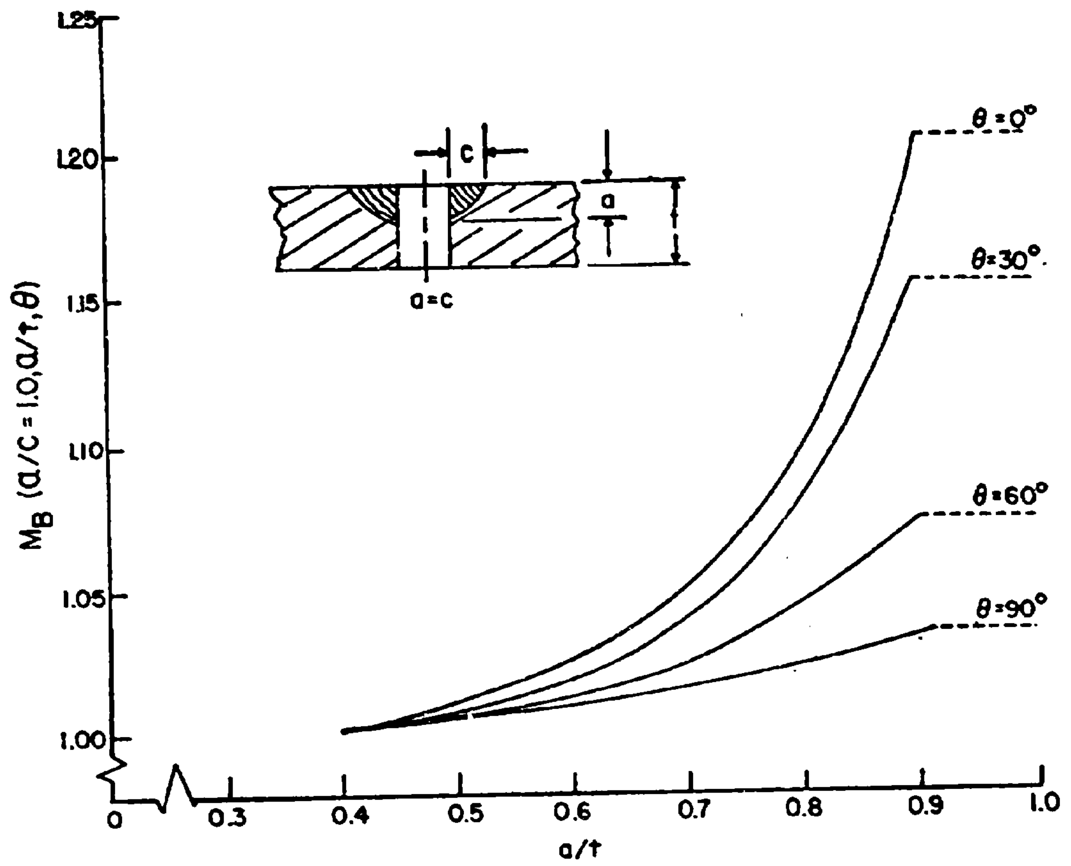


Figure 4.30 Assumed Back Surface Stress Intensity Magnification Factor for Quarter-Circular Part-Through Fastener Hole Flaws

$$K_{I_e} = \sigma\sqrt{\frac{\pi a}{Q}}\left[\cos^2\theta + \frac{a^2}{c^2}\sin^2\theta\right]^{\frac{1}{4}} \quad a/c \le 1.0$$

$$K_{I_e} = \sigma\sqrt{\frac{\pi c}{Q}}\left[\sin^2\theta + \frac{c^2}{a^2}\cos^2\theta\right]^{\frac{1}{4}} \quad a/c > 1.0$$

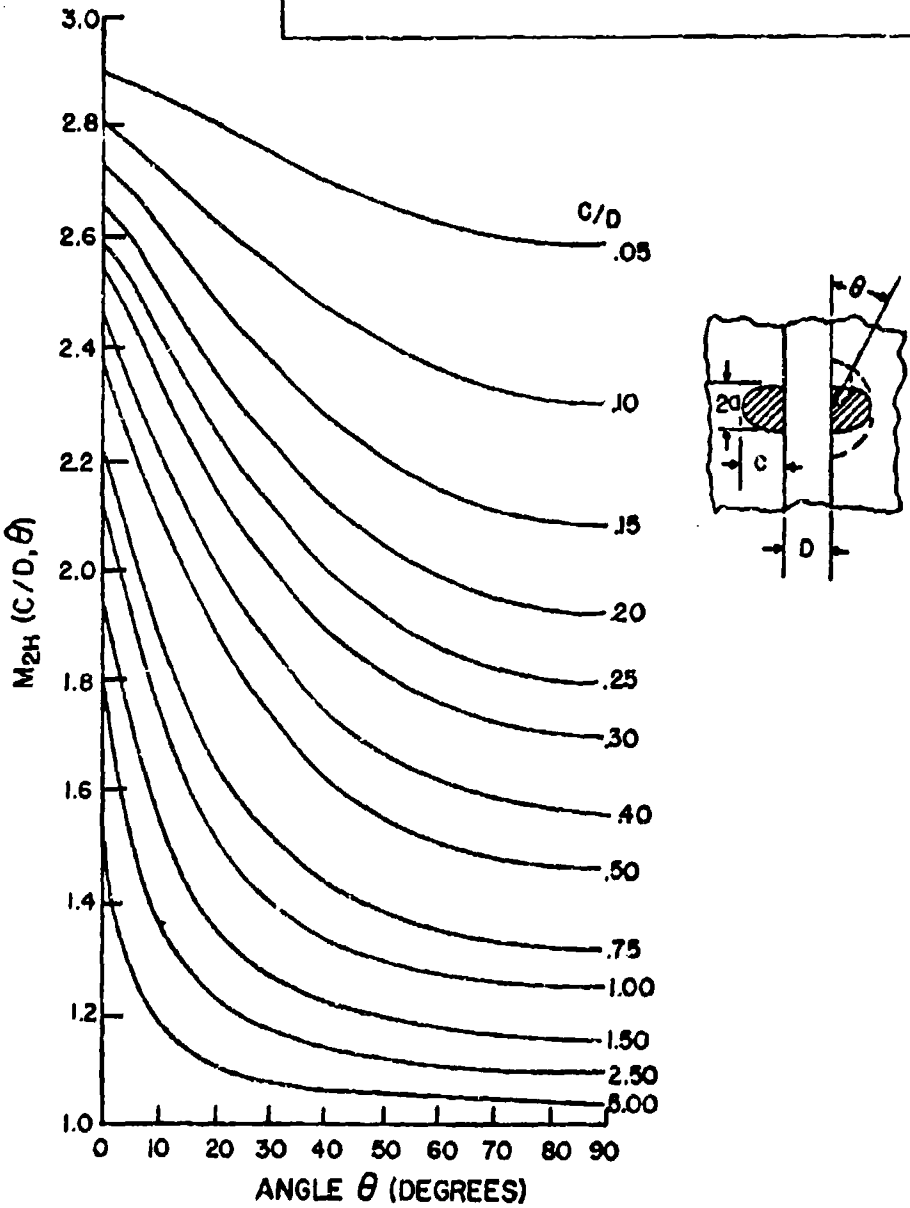


Figure 4.31 Correction Factor for Two Embedded Elliptical Flaws at a Hole

$$K_{Ieb} \approx \sigma_b \sqrt{\frac{\pi a}{Q}} \left[\cos^2\theta + \frac{a^2}{c^2}\sin^2\theta\right]^{\frac{1}{4}} \qquad a/c \le 1.0$$

$$K_{Ieb} = \sigma_b \sqrt{\frac{\pi c}{Q}} \left[\sin^2\theta + \frac{c^2}{a^2}\cos^2\theta\right]^{\frac{1}{4}} \qquad a/c > 1.0$$

σ_b = Bearing Stress = P/(2rt)

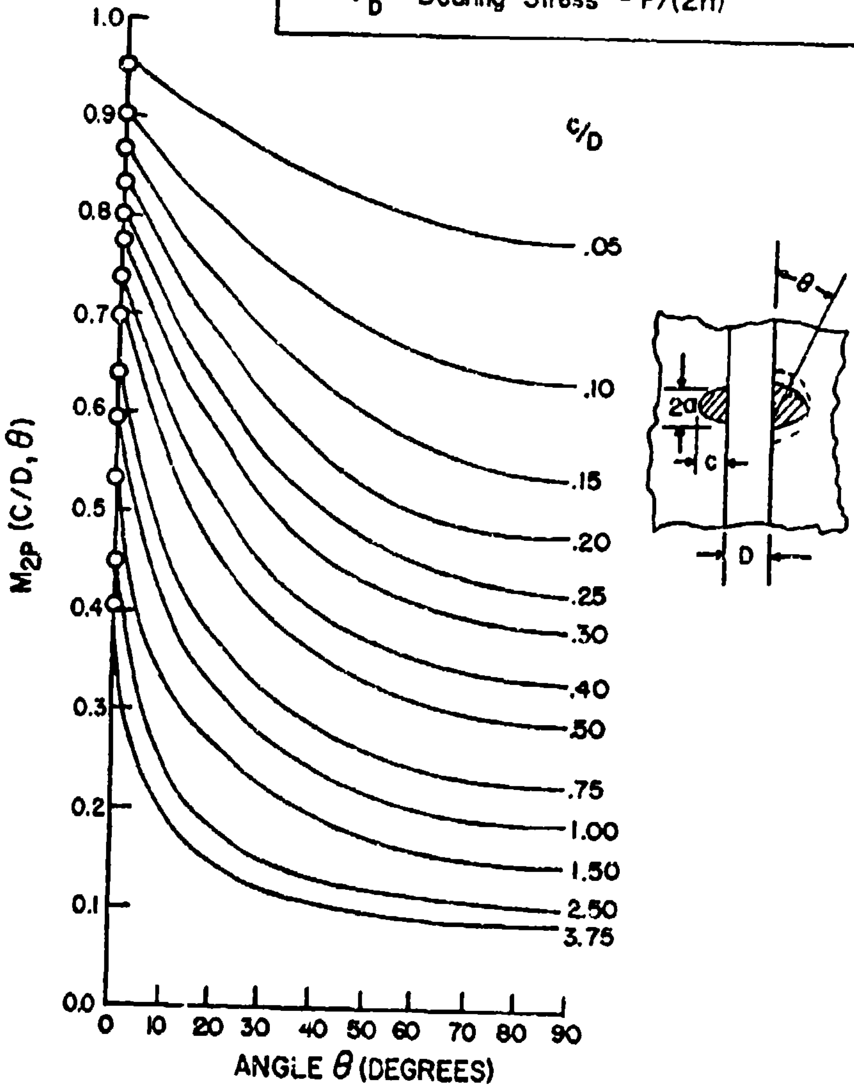


Figure 4.32 Correction Factor for Two Embedded Elliptical Flaws at a Pin-Loaded Hole

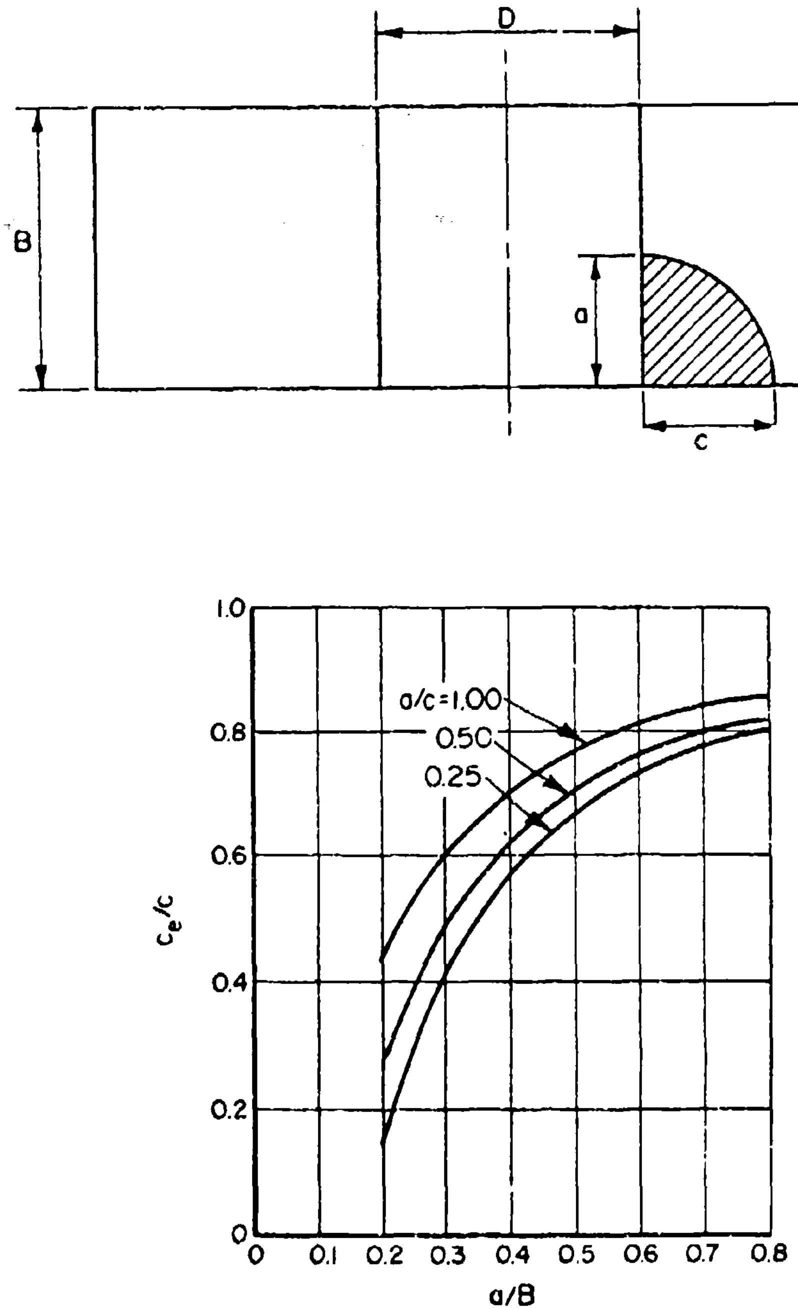


Figure 4.33 Empirical Curves to Determine the Effective Size of a Corner Crack at a Hole

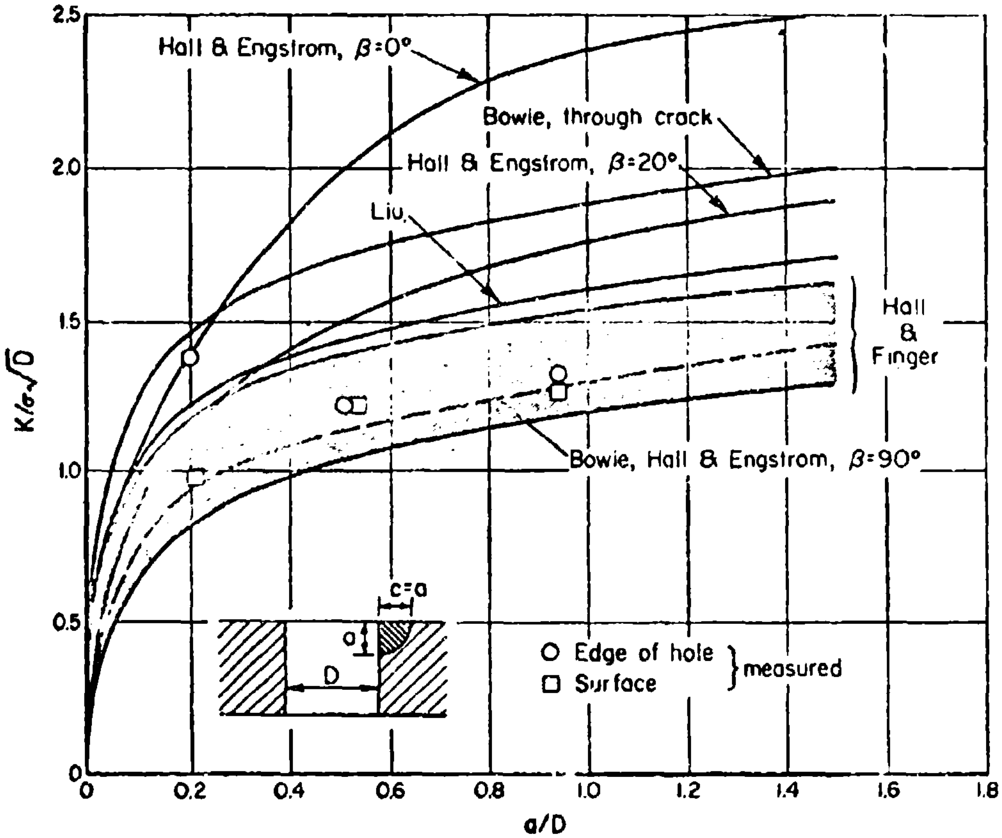


Figure 4.34 Comparison of Approximate Solutions for the Stress-Intensity Factor of a Quarter-Circular Corner Crack at the Edge of an Open Hole

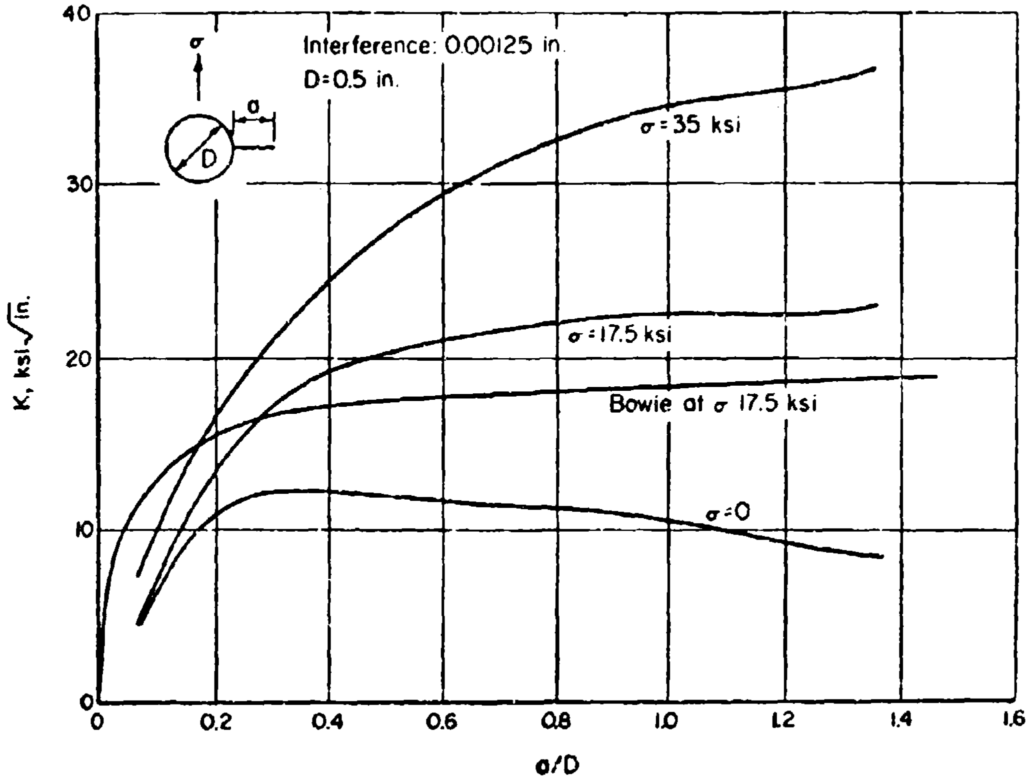


Figure 4.35 Stress-Intensity Factor for Through-Cracked Hole with Interference Fit Fastener

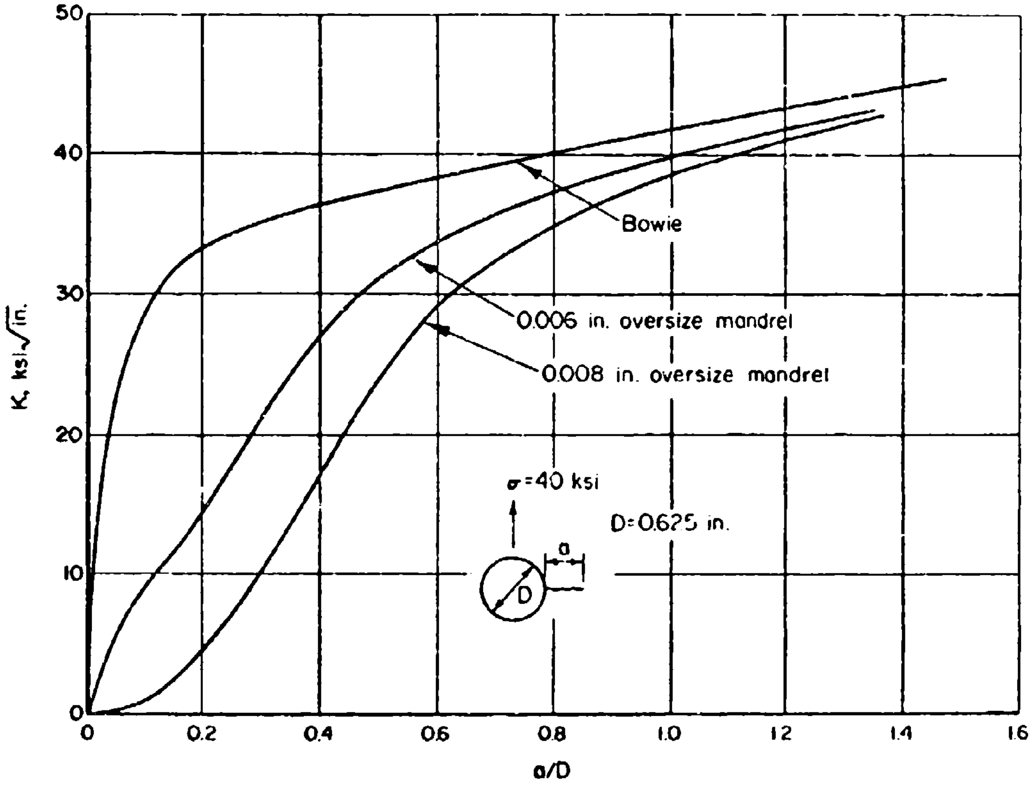


Figure 4.36 Stress-Intensity Factor for Through Cracks at Cold Worked Holes

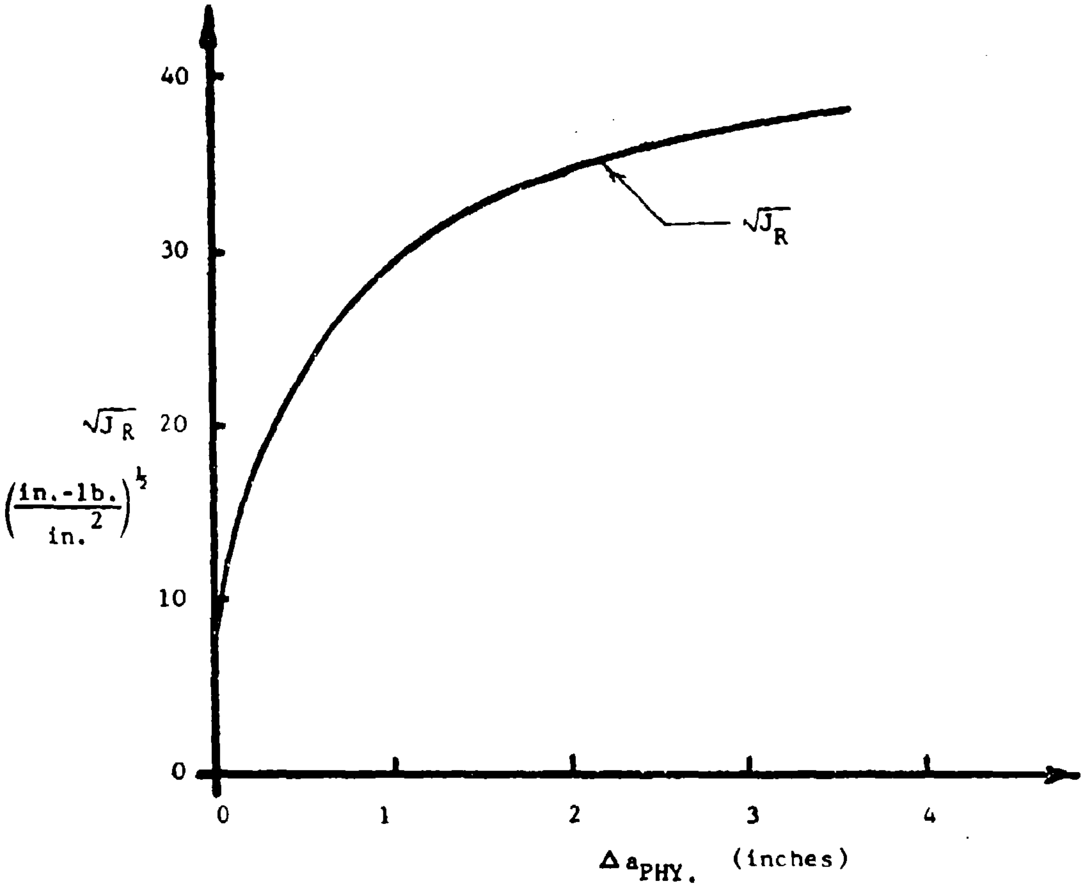


Figure 4.37 Square Root of J_R Resistance Curve

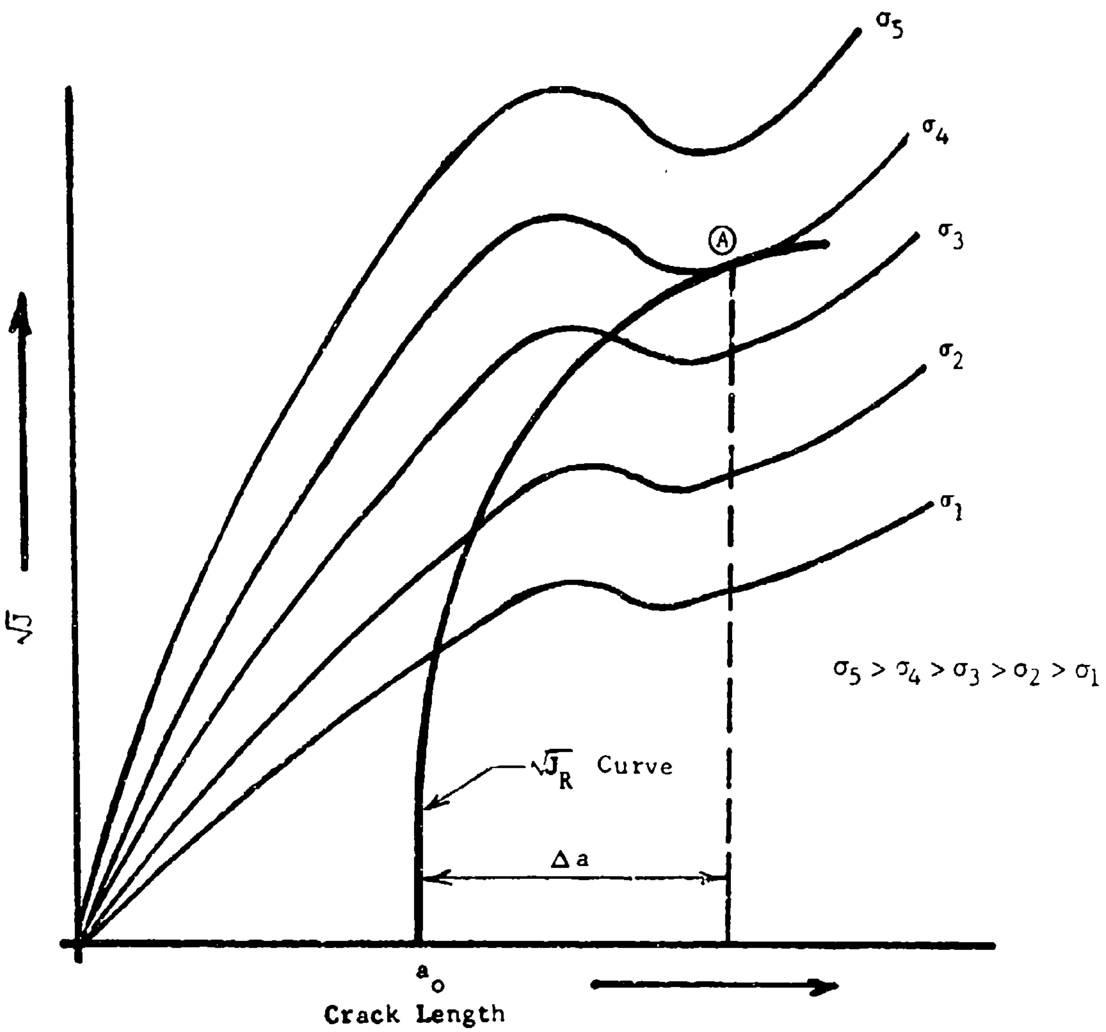


Figure 4.38 Failure Analysis Based on $J_{Critical}$ Curve

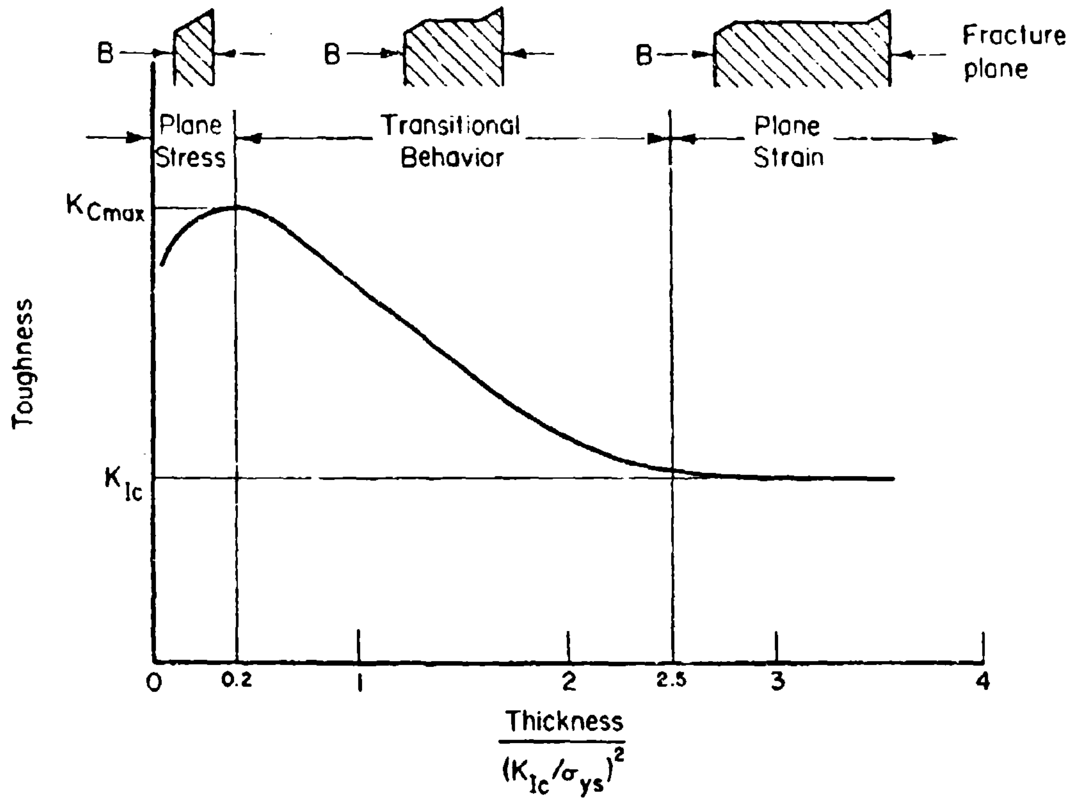


Figure 4.39 Diagrammatic Dependence of Toughness on Thickness

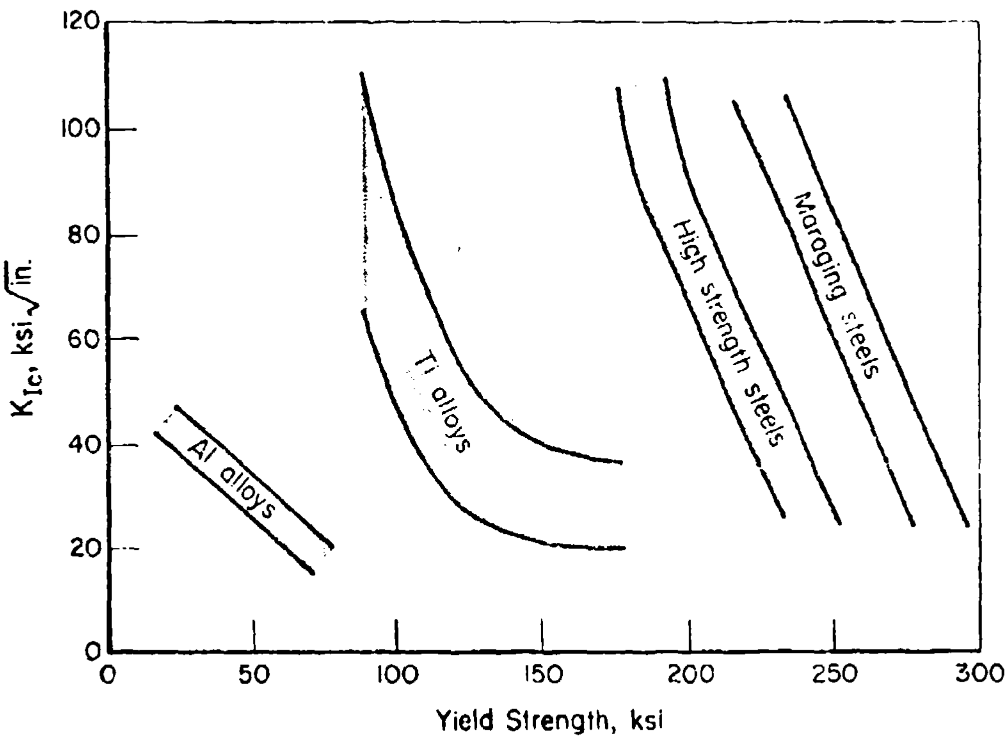


Figure 4.40 Fracture Toughness as a Function of Yield Stress

TABLE KF82 AVERAGE PLANE-STRAIN FRACTURE-TOUGHNESS DATA FOR AISI 4340 ALLOY STEEL PLATE

ALLOY	CONDITION	FORM AND THICK IN.	TEST TEMP F	AVG YIELD STR KSI	SPEC DESIGN	SPEC ORIENT	SPECIMEN DIMENSIONS THICK (B) IN.	WIDTH (W) IN.	CRACK LENGTH IN.	AVG KIC KSI-SQRT (IN.)	MIN KIC KSI-SQRT (IN.)	NO. OF SPEC	2.5 (KIC/YIELD) IN.	REF
AISI 4340 (VIM) STEEL 89	1600 F 0.5 HR,AC, 1500 F 0.5 HR,OQ, 400 F 4 HR,AC, 250 F 24 HR,AC	P 0.5	70	224	BEND	T-L	0.5	1.0		76		2	0.29	81435
AISI 4340 (VIM) STEEL 93	1500 F 0.5 HR,AC, 1500 F 0.5 HR,OQ, -100 F 0.5 HR, 400 F 4 HR,AC, 250 F 24 HR,AC	P 0.5	70	231	BEND	T-L	0.5	1.0		64		2	0.22	81435
AISI 4340 (CODE 1)	1550 F,OQ,TEMPERED 500 F	P 1.0	70	238	BEND	L-T	0.95	2.0	1.0	44.4	40.7	35	0.09	83833
AISI 4340(CODE [illegible])	1550F,OQ,500F 2+2HR	P 0.4	70	217	BEND	L-T	0.39	0.97	0.34-0.44	52.5	50.9	3	0.15	7317
AISI 4340(CODE [illegible])	SAME	P 0.4	70	217	BEND	L-T	0.37	0.99	0.20-0.35	50.5	49.9	3	0.13	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	70	226	BEND	L-T	0.37	0.99	0.21-0.43	55.9	51.4	3	0.15	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	200	208	BEND	L-T	0.39	0.95	0.31-0.43	59.9	56.9	3	0.21	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	200	205	BEND	L-T	0.37	0.99	0.20-0.34	57.3	55.2	3	0.20	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	-50	231	BEND	L-T	0.39	0.99	0.15-0.37	42.1	41.2	3	0.09	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	-50	231	BEND	L-T	0.37	0.98	0.20-0.35	37.6	37.4	3	0.06	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	-50	235	BEND	L-T	0.37	0.99	0.20-0.35	39.8	38.2	3	0.07	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	-100	235	BEND	L-T	0.39	0.99	0.33-0.44	35.9	35.1	3	0.06	73172
AISI 4340(CODE [illegible])	SAME	0.4	-100	235	BEND	L-T	0.37	0.98	0.20-0.31	31.4	29.1	3	0.04	73172
AISI 4340(CODE [illegible])	SAME	P 0.4	-100	240	BEND	L-T	0.37	0.99	0.20-0.34	33.1	30.6	3	0.05	73172
AISI 4340 (CODE [illegible])	1550 F,OQ,TEMPERED [illegible] F	P 1.0	70	206	BEND	L-T	0.95	2.0	1.0	79.6	71.5	37	0.38	83833
AISI 4340	1550 F,OQ,TEMPERED TO 140-[illegible] KSI YS	P [illegible]	75	149	BEND		3.0	6.0	3	133		3	2.3	[illegible]

COMPOSITIONS (WEIGHT PERCENT)

	C	MN	P	S	SI	NI	CR	MO	CO	V	CU	AL	CA	N	O
STEEL 89	0.39	0.74	0.001	0.0005	0.29	1.95	0.96	0.24		0.005		0.025		0.002	0.001
CODE 1	0.42	0.71	0.010	0.017	0.25	1.77	0.09	0.23							
CODE 16	0.43	0.96	0.006	0.005	0.27	1.79	0.74	0.24							
CODE 19	0.41	0.59	0.019	0.012	0.29	1.97	0.79	0.22							
CODE 10	0.40	0.75	0.0-5	0.011	0.29	1.77	0.79	0.25							

Figure 4.41 Example of K_{Ic} Data Presentation in Damage Tolerant Design Handbook

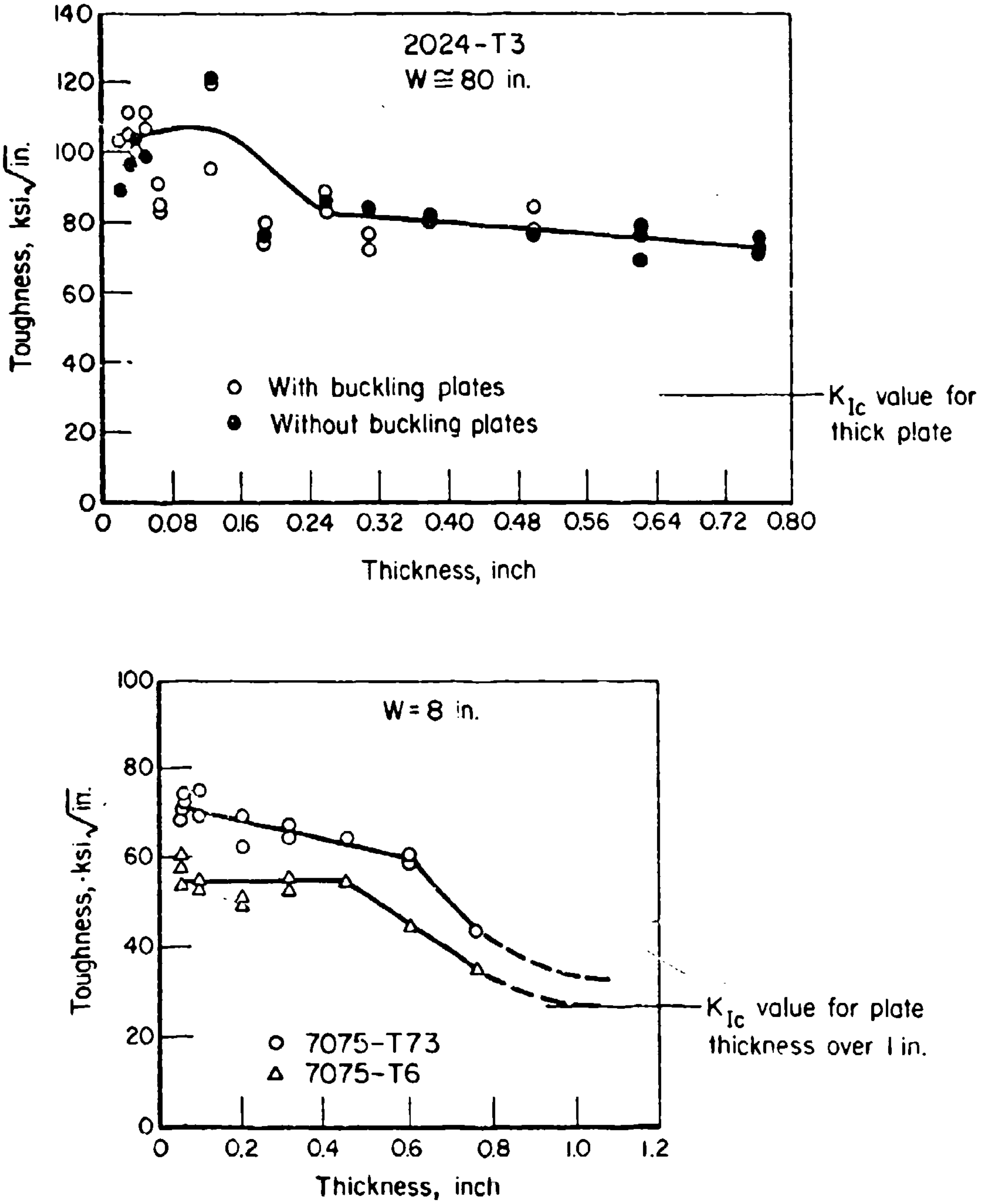


Figure 4.42 Toughness as a Function of Thickness

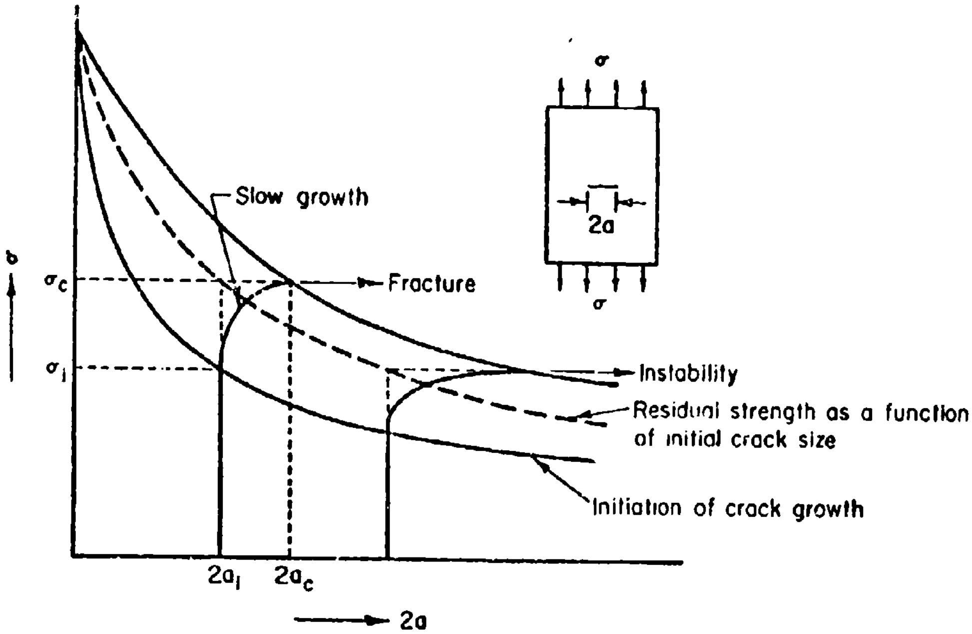
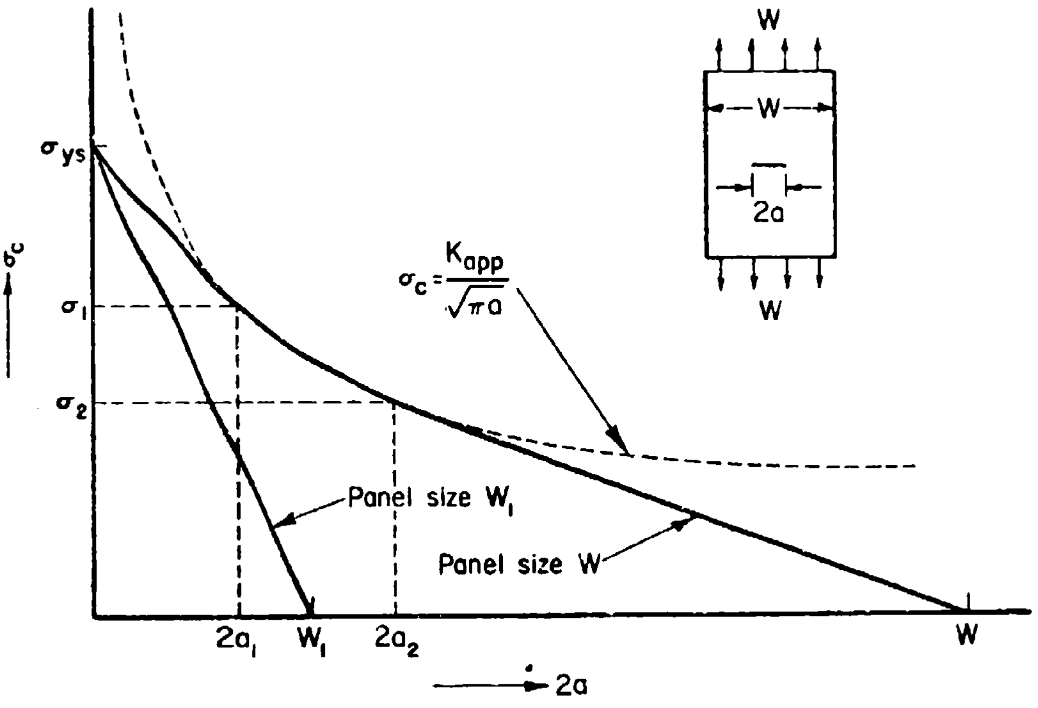
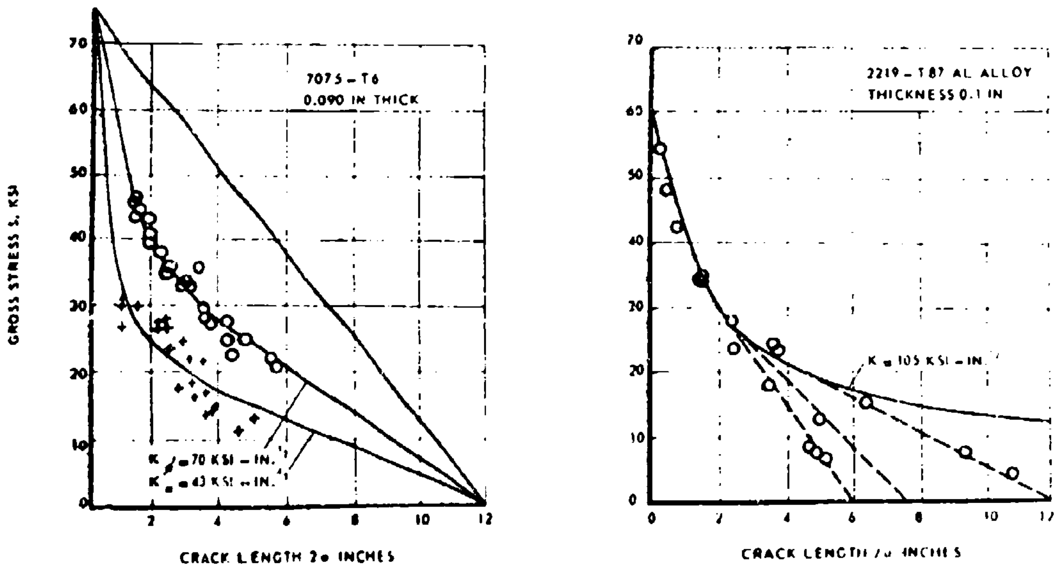


Figure 4.43 Residual-Strength Behavior in Plane Stress and Transitional States

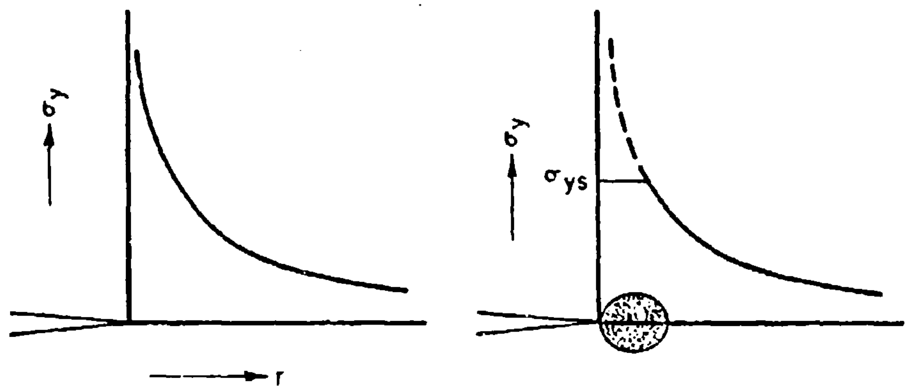


a. Feddersen's approach



b. Test data showing agreement with Feddersen's approach

Figure 4.44 Plane-Stress Fracture Toughness



a. Elastic stress distribution

b. Idealized elastic-plastic stress distribution

Figure 4.45 Axial Stress at Crack Tip and Plastic Zone

CHAPTER 5

Analysis of Damage Growth

CHAPTER 5. CRACK PROPAGATION

LIST OF FIGURES

FIGURE

LIST OF FIGURES (CONTINUED)

FIGURE

LIST OF FIGURES (CONTINUED)

FIGURE

5.1 BASIC INFORMATION

5.1.1 Introduction

MIL-A-83444 "Airplane Damage Tolerance Design Requirements," specifies that cracks shall be assumed to exist in all primary aircraft structure. These cracks shall not grow to a size to cause loss of the aircraft at a specified load within a specified period. Showing compliance with these requirements implies that the rate of growth of the assumed flaws must be predicted.

Crack growth is a result of cyclic loading due to maneuvers and gusts (fatigue cracking), or of combined action of stresses and environment (stress-corrosion cracking), or both. The most common crack-growth mechanisms are fatigue-crack growth and environment-assisted fatigue-crack growth. Certain aircraft parts (especially high-strength forgings) may be liable to stress-corrosion cracking. Since there is a design threshold for stress corrosion, proper detail design and proper material selection can minimize or prevent stress corrosion. Fatigue cracking is difficult to prevent, but it can be controlled.

FRACTURE-MECHANICS CONCEPTS ALLOW THE PREDICTION OF CRACK GROWTH ON THE BASIS OF THE SAME PARAMETER AS USED FOR RESIDUAL-STRENGTH PREDICTION, NAMELY THE STRESS-INTENSITY FACTOR. Consequently, all the information concerning the determination of stress-intensity factors, as presented in Chapter 4, applies equally to crack-growth analysis.

In principle, the prediction of crack growth requires the following steps:

(1) Determine the stress-intensity factor as a function of crack size for the relevant crack geometry and the relevant structural geometry.

(2) Establish the (cyclic) stress-time history for the structure or component under consideration.

(3) Find the baseline crack-growth properties (crack-growth rate as a function of the stress-intensity factor) for the material used in the design and for the relevant environment.

(4) Integrate the crack-growth rate [from (3)] to a crack-growth curve, using the proper stress-time history [from (2)], the proper stress-intensity formulation [from (1)], and an appropriate integration rule. A typical crack-growth curve is shown in Figure 5-1.

CRACK-GROWTH CALCULATIONS ARE MORE COMPLEX THAN RESIDUAL-STRENGTH CALCULATIONS, BECAUSE MORE FACTORS HAVE TO BE CONSIDERED. This chapter provides guidelines to arrive at best estimates possible, and points out where deficiencies in knowledge and analysis methods lead to inaccuracies.

5.1.2 Fatigue-Crack Growth and Stress Intensity

Consider constant-amplitude fatigue loading as in Figure 5-2. The following parameters are defined:

σ_m	mean stress
σ_a	stress amplitude
$\Delta\sigma$	stress range
σ_{max}	maximum stress
σ_{min}	minimum stress
R	stress ratio: $R = \dfrac{\sigma_{min}}{\sigma_{max}} = \dfrac{\sigma_m - \sigma_a}{\sigma_m + \sigma_a} = 1 - \dfrac{\Delta\sigma}{\sigma_{max}}$.

The cyclic stress can be fully characterized (apart from the frequency) by any combination of two of these parameters. The stress range, $\Delta\sigma$, and the stress ratio, R, are the two most commonly used. Note that in a constant-amplitude test each of these parameters has a constant value with respect to time.

The stress history can be converted into a stress-intensity history (Figure 5-2). The following parameters are defined:

K_a	amplitude of the stress intensity $= \beta\sigma_a\sqrt{\pi a}$
ΔK	range of the stress intensity $= \beta\Delta\sigma\sqrt{\pi a}$
K_{max}	maximum stress intensity $= \beta\sigma_{max}\sqrt{\pi a}$
K_{min}	minimum stress intensity $= \beta\sigma_{min}\sqrt{\pi a}$
R	cycle ratio $R = \dfrac{K_{min}}{K_{max}} = 1 - \dfrac{\Delta K}{K_{max}} = \dfrac{\sigma_{min}}{\sigma_{max}}$.

The stress-intensity history can be characterized by any combination of two of these parameters. The two most commonly used are ΔK and R. Note that crack length increases during crack growth. Consequently, all stress-intensity parameters increase during crack growth under constant amplitude loading. However, the stress ratio, R, is constant.

In the elastic case, the stress-intensity factor is a sufficient parameter to describe the whole stress field at the tip of a crack. When the plastic zone at the crack tip is small compared with the crack size, the stress-intensity factor still gives a good indication of the stress environment of the crack tip. IF TWO DIFFERENT CRACKS HAVE THE SAME STRESS ENVIRONMENT (EQUAL STRESS-INTENSITY FACTORS), THEY BEHAVE IN THE SAME MANNER AND SHOW THE SAME RATE OF GROWTH. SINCE TWO PARAMETERS ARE REQUIRED TO CHARACTERIZE THE CYCLE, TWO PARAMETERS ARE REQUIRED TO CHARACTERIZE CRACK GROWTH.

The crack-growth rate per cycle, da/dN, where N is the cycle number, can be given as

$$\frac{da}{dN} = f\ (\Delta K,\ R) = g\ (\Delta K,\ K_{max}) \quad . \qquad (5.1)$$

In the case that $\sigma_{min} = 0$ (i.e., $R = 0$ and $\Delta K = K_{max}$), the expression reduces to

$$\frac{da}{dN} = f\ (\Delta K) \quad . \qquad (5.2)$$

Consequently, the rate of growth of a 2-inch crack at $\Delta\sigma = 10$ ksi and $R = 0$ is equal to the growth rate of a 0.5-inch crack at $\Delta\sigma = 20$ ksi, $R = 0$ (assume $\beta \approx 1$).

5.1.3 Crack-Growth Equations

Figure 5-3 shows an example of da/dN data as a function of ΔK for different R ratios. Obviously f (ΔK, R) in Equation (1) is not a simple function. Many equations have been proposed (1-4) for it, but none of them is of general validity. THESE EQUATIONS ARE NOT LAWS. THEY ARE MERELY MATHEMATICAL REPRESENTATIONS OF CRACK-GROWTH BEHAVIOR FOR THE PURPOSE OF INTEGRATION. Only the most general equations will be presented in this section.

Typically, the plot of da/dN versus ΔK is made on double-logarithmic paper (Figure 5-3). On these scales, the resulting curves are usually S shaped. Sometimes a straight line portion is found for the central region, which for $R = 0$ would lead to

$$\frac{da}{dN} = C\ \Delta K^n \tag{5.3}$$

Although Equation (5.3) can sometimes be used as a rule of thumb, it is not a general equation.

The S shape of the curves suggests that there are two asymptotes. The one at the high ΔK is governed by the final failure conditions. If the maximum stress intensity is equal to the fracture toughness ($K_{max} = K_c$ or $K_{max} = K_{Ic}$), fracture will occur. Hence, da/dN approachs ∞, if K_{max} approaches K_{Ic}. This behavior is reflected in the equation proposed by Forman (5),

$$\frac{da}{dN} = \frac{C\ \Delta K^n}{(1-R)\ K_{Ic} - \Delta K} = \frac{C\ \Delta K^n}{(1-R)\ (K_{Ic} - K_{max})} \quad , \tag{5.4}$$

which can be rearranged to give

$$\frac{da}{dN} = \frac{C\ \Delta K^{n-1} K_{max}}{K_{Ic} - K_{max}} \quad . \tag{5.5}$$

NOTE: ALTHOUGH THE TERM K_{Ic} IS USED IN EQUATIONS (5.4) AND (5.5), THE APPROPRIATE CRITICAL VALUES FOR THE APPROPRIATE MATERIAL THICKNESS WILL GENERALLY HAVE TO BE USED FOR DATA ANALYSIS.

The asymptote at low ΔK is associated with a threshold value of the stress intensity below which a crack would be nonpropagating. There is no concurrence of opinion as to the uniqueness of a threshold. Experimental measurements at extremely slow growth rates are difficult and the results can be deceiving. Where sufficient data are available, the threshold can be accounted for in the sigmoidal equation proposed by Collipriest (6,7):

$$\frac{da}{dN} = \exp\left[n \cdot \frac{\ln K_c - \ln \Delta K_{th}}{2} \cdot \tanh^{-1}\left\{\frac{\ln \Delta K - \dfrac{\ln(1-R)K_c + \ln \Delta K}{2}}{\dfrac{\ln(1-R)K_c - \ln \Delta K_{th}}{2}}\right\} + \ln\left\{C \cdot \exp\left(n \cdot \frac{\ln K_c - \ln K_{th}}{2}\right)\right\}\right] \tag{5.6}$$

which was modified by Davies and Feddersen (8) into:

$$\log_{10}\frac{da}{dn} = C_1 + C_2 \operatorname{arctanh} \frac{\log_{10} \dfrac{K_c K_{th}}{\left[K_{max}(1-R)^m\right]^2}}{\log_{10} \dfrac{K_{th}}{K_c}} \quad . \qquad (5.7)$$

In these equations, ΔK_{th} is the threshold ΔK; whereas, K_{th} is the threshold K_{max}. The sigmoidal shape of the equations accounts for two asymptotes at K_c and K_{th}. The use of K_c in Equations (5.6) and (5.7) is subject to the same restriction as is the use of K_{Ic} in Equation (5.5) (see note above).

EACH OF THE CRACK-GROWTH EQUATIONS CAN GIVE A REASONABLE REPRESENTATION OF CRACK-GROWTH DATA WITHIN CERTAIN LIMITATIONS. Equation (5.3), for example, will give an approximation in the central region of the da/dn - ΔK diagram. The most general equation is probably Equation (5.7) which was shown (8) to give the best correlation with large sets of data (Figure 5-4).

An equation for the da/dn - ΔK diagram is useful when it can be readily integrated for simple manual calculations. Therefore, the expression $da/dn = C(\Delta K)^n$ is of value for a quick but rough appraisal of crack growth.

A general equation should include the effect of stress ratio. Apart from the two equations mentioned above, several other equations have been proposed to account for the stress ratio effect. The Forman equation is most widely used, because of its simplicity and because it is expressed in ΔK, rather than in an effective ΔK.

For computer applications an equation is not really necessary, since graphical or tabular data are equally convenient for computer use as an equation.

The Damage Tolerance Data Handbook (9) provides crack-growth data for a variety of materials. The data are presented in the form of graphs, an example of which is given in Figure 5.5. No equation fitting was attempted. The raw data can be readily used for computer integration, taking into account the guidelines given in this volume.

NO EQUATION FITTING SHOULD BE ATTEMPTED IF ONLY LIMITED SETS OF DATA ARE AVAILABLE. In case limited data sets have to be used, a comparison should be made with similar alloys for which complete data are available, and curves may be fitted through the limited data sets on the basis of this comparison.

5.1.4 Factors Affecting Crack Growth

Unlike tensile strength and yield stress, fatigue-crack-propagation behavior is not a consistent material characteristic. Fatigue-crack growth is influenced by many uncontrollable factors. As a result, a certain amount of scatter occurs. THEREFORE, CRACK-GROWTH PREDICTIONS SHOULD BE BASED ON FACTORS RELEVANT TO THE CONDITIONS IN SERVICE.

Among the many factors that affect crack propagation, the following should be taken into consideration for crack-growth predictions:

A
- Type of product (plate, extrusion, forging)
- Heat treatment
- Orientation with respect to grain direction
- Manufacturer and batch
- Thickness

B
- Environment
- Temperature
- Frequency.

No attempt will be made to illustrate the effects of all these factors with data, particularly because some factors have largely different effects on different materials. Rather, some general trends will be briefly mentioned.

The factors under A pertain to the material. The crack-propagation characteristics for a particular alloy differs for plates, extrusions, and forgings. The latter may exhibit a rather large anisotropy, which may have to be considered in the growth of surface flaws and corner cracks, which grow simultaneously in two perpendicular directions. Closely related to this are the other processing variables, particularly the heat treatment.

An alloy of nominally the same composition but produced by different manufacturers may have largely different crack-propagation properties (10). This is illustrated in Figure 5-6. The differences are associated with slight variations in composition, inclusion content, heat treatment (precipitates), and cold work. Similar variations in crack growth occur for different batches of the same alloy produced by the same manufacturer.

There is a small but systematic effect of thickness on crack propagation (11-15). Some data are presented in Figure 5-7, showing that growth rates are higher in thicker sheets.

IN VIEW OF THESE VARIATIONS IN CRACK-GROWTH PROPERTIES, PREDICTIONS OF CRACK GROWTH SHOULD BE BASED ON MATERIAL DATA FOR THE RELEVANT PRODUCT FORM, THE RELEVANT THICKNESS. SPOT CHECKS MAY BE NECESSARY TO ACCOUNT FOR VARIABILITIES IN HEATS AND/OR MANUFACTURER.

The factors under B are associated with the environmental circumstances. A lightly corrosive environment (humid air) gives rise to higher crack-growth rates than a dry environment.(16-25). This effect is illustrated in Figure 5-8. Although there is no concurrence of opinion as to the explanation of the environmental effect, it is certainly due to corrosive action. As a result, the influence of the environment is time and temperature dependent. Therefore, it is usually assumed that the small but systematic effect of cycling frequency(17,20,24,26) is related to the environmental effect.

At low temperatures the reaction kinetics are slower and the air can contain less water vapor. This may reduce crack-propagation rates in certain alloys(27) (e.g., Figure 5-9). Sometimes the effect of temperature on fatigue-crack growth in the low temperature range is very small(28). Temperatures higher than ambient may increase crack-growth rates(29,30).

THE SIGNIFICANT EFFECT OF ENVIRONMENT ON CRACK GROWTH SHOULD BE TAKEN INTO ACCOUNT FOR CRACK-GROWTH PREDICTIONS. CRACK-GROWTH DATA SHOULD BE USED THAT REPRESENT THE EFFECT OF THE EXPECTED ENVIRONMENT AND TEMPERATURE.

5.1.5 Data Acquisition; Use of Data

Fatigue-crack-propagation data for a variety of materials can be found in data handbooks. In many cases, however, the data for a particular application (with regard to material condition, thickness, and environment) will have to be generated. In principle this could be done for any kind of specimen for which a stress-intensity solution is known, so that the crack-growth rate could be determined as a function of ΔK. Recently, compact tension specimens have been used extensively for this purpose. HOWEVER, THE COMPACT TENSION SPECIMEN SHOULD NOT BE USED FOR R RATIOS LESS THAN ZERO, BECAUSE THE STRESS DISTRIBUTION UNDER COMPRESSIVE LOAD IS NOT REPRESENTATIVE FOR SERVICE LOADING. CENTER-CRACKED PANELS OR EDGE-NOTCHED PANELS ARE MORE RELEVANT TO AIRCRAFT APPLICATIONS AND SINCE $R < 0$ IS A RELEVANT CASE FOR AIRCRAFT STRUCTURES, THEIR USE IS RECOMMENDED. SURFACE FLAW SPECIMENS ARE USEFUL FOR SPECIAL APPLICATION.

THEY ARE NOT RECOMMENDED FOR THE GENERATION OF GENERAL PURPOSE BASELINE DATA.

The first part of the crack extension should be discarded from the data (e.g., approximately the first 0.05 inch for a center crack) since it contains the variability of initiation and early crack growth as affected by the notch machining procedures. Therefore, it cannot be generalized. The crack length is defined as the total damage size (i.e., notch plus physical crack).

The range of the stress-intensity factor is determined using the relevant formula for $\Delta K = \beta \, \Delta\sigma \sqrt{\pi a}$, and by substituting the proper values for $\Delta\sigma$ and a. The crack-growth rate is determined as an average over a small amount of crack growth (in the order of 0.02 = 0.04 inch). A generally accepted procedure is to take an average of 2 successive crack increments by a 3-point divided difference method (8). Suppose the data are

$$a_i - N_i$$

$$a_j - N_j$$

$$a_k - N_k \quad ,$$

where N is the cycle number at a crack size a_i, etc. The crack-growth rate at crack size a_j is then

$$\left.\frac{da}{dN}\right|_{a_j} = \frac{a_j - a_i}{N_j - N_i} + \frac{N_j - N_i}{N_k - N_i} \quad \frac{a_k - a_j}{N_k - N_j} - \frac{a_j - a_i}{N_j - N_i} \tag{5.8}$$

The crack-growth records usually contain slight irregularities as a result of either local differences in material behavior or inaccuracies in crack measurements. A hypothetical example is shown in Figure 5-10a. The outlying data points are indicated by an asterisk. In the da/dN - ΔK plot these irregularities will also show up as indicated in Figure 5-10b. If more tests are run and all the data compiled, the plot will be as in Figure 5-10c: each test might have a few outlying data points, but the compilation has many outlying points.

If these outlying data points were considered real, the results whould show the wide apparent scatter band as shown in Figure 5-10c. However, as can be seen from Figure 5-10a, these points did not affect the crack-growth curve. Therefore, they should be discarded from the analysis. This would result in the much narrower real scatter band shown in Figure 5-10c. If the wide scatter band was considered for a crack-growth prediction, the upper bound would predict a consistent high growth rate at each crack size (whereas it happened only incidentally as shown in Figure 5-10a). As a result, the diagram would reflect a large apparent scatter in crack-growth lives (Figure 5-10d) whereas the real scatter in crack growth lives appears to be smaller.

IT CAN BE CONCLUDED THAT IT WOULD NOT BE AN OBJECTIONABLE PROCEDURE TO FIT A SMOOTH CURVE THROUGH THE a-N DATA AND TO DIFFERENTIATE THIS CURVE TO OBTAIN $da/dN - \Delta K$. THIS WOULD AUTOMATICALLY DISCARD OUTLYING DATA POINTS. THE REAL SCATTER AS IN FIGURE 5-10d STILL WOULD BE REFLECTED, PROVIDED 3 OR 4 TESTS WERE USED FOR THE ANALYSIS. FOR A REASONABLY CONSERVATIVE CRACK-GROWTH PREDICTION, THE UPPER BOUND OF THE REAL SCATTER BAND SHOULD BE USED.

As mentioned already in Section 5.1.3, the data at the upper end of the $da/dN - \Delta K$ curve have to be discarded if the panels are too small to yield a valid K_c for the thickness under consideration. At high crack-growth rates, crack extension is largely determined by the failure properties of the specimen (i.e., K_c in the case of wide panels). Small panels, however, do not fail at K_c as was discussed extensively in Section 4.3.1. Rather, they fail approximately at net section yield, which is at an apparent K_c much lower than the real K_c. This behavior is reflected in fatigue-crack growth.

As an example, consider a material with $K_c = 100$ ksi$\sqrt{\text{in.}}$ and a yield strength of 60 ksi. According to Equation (4.67) of Chapter 4, the required panel size for a valid K_c is 12 inches. Smaller panels will fail at $K < K_c$ by net section yield. Figure 5-11a shows how this affects the upper end of the $da/dN - \Delta K$ curve for various specimen sizes. Panels of 12-inch width and larger would show the general behavior.

By the same token, the upper end is dependent upon the test stress level in the case of small specimens. The crack size at failure depends upon stress level, whereas the apparent K_c depends upon crack size. As a result, the da/dN curve is affected as shown in Figure 5-11b.

A GENERAL $da/dN - \Delta K$ CURVE CAN BE ESTABLISHED ONLY WITH A SPECIMEN OF SUFFICIENT SIZE. FOR SMALL SPECIMENS THE UPPER END OF THE CURVE WILL REFLECT SPECIMEN BEHAVIOR RATHER THAN MATERIAL BEHAVIOR. CONSEQUENTLY, TEST DATA OBTAINED FROM SMALL SPECIMENS SHOULD NOT BE USED FOR CRACK-GROWTH ANALYSIS OF LARGE PANELS AND VICE VERSA, UNLESS THE UPPER PART OF THE CURVE IS APPROPRIATELY MODIFIED.

5.1.6 Stress-Corrosion Cracking

For a given material-environment interaction, the stress-corrosion-cracking rate is also governed by the stress-intensity factor. Similar specimens with the same size of initial crack but loaded at different levels (different initial K values) show different times to failure[31-33] as shown in Figure 5-12. A specimen initially loaded to K_{Ic} failes immediately. The threshold level is denoted as K_{Iscc}.

If the load is kept constant during the stress-corrosion-cracking process, the stress intensity will gradually increase due to the growing crack. As a result the crack-growth rate per unit of time, da/dt, increases according to

$$\frac{da}{dt} = f(K) \quad . \tag{5.9}$$

When the crack has grown to a size that K becomes equal to K_{Ic}, the specimen fails. This is shown diagrammatically in Figure 5-13. In a typical test the specimen is loaded to a given initial K. The time to failure is recorded, giving rise to the typical data point shown in Figure 5-13. During the test K will increase (as a result of crack extension) from its initial value to K_{Ic}, where final failure occurs. This is reflected by the schematic crack growth curves in Figure 5-13.

The stress-corrosion threshold and the rate of growth depend upon the material and the environmental conditions. Data on K_{Iscc} and da/dt can be found in the Damage Tolerance Data Handbook[9]. A typical example of data presentation is shown in Figure 5-14. As illustrated in Figure 5-15, a component with a given crack fails at a stress given by $\sigma_c = K_{Ic}\ \phi\beta\sqrt{\pi a}$. It will show stress-corrosion-crack growth when loaded to stresses in excess of $\sigma_{scc} = K_{Iscc}/\beta\sqrt{\pi a}$.

In service stress-corrosion cracks have been found to be predominantly a result of residual stresses and secondary stresses. Stress-corrosion failures due to primary loading seldom occur. This is partly due to the fact that most stress-corrosion cracks occur in the short transverse direction which is usually not the primary load direction. In many materials the long transverse and longitudinal directions are not very susceptible to stress corrosion. However, if stress corrosion can occur it will have to be accounted for in damage tolerance analyses.

In principle the crack-propagation curve for stress corrosion can be obtained from an integration of Equation (5.9). However, reliable da/dt are very scarce. Also, the best design policy to handle stress corrosion cracking is to prevent it, rather than controlling its growth as done for fatigue cracking. This would, therefore, mean that stress-corrosion critical components be designed to operate at a stress level lower than $\sigma_{scc} = K_{Iscc}/\beta\sqrt{\pi a_i}$ in which a_i would be the assumed initial flaw sizes as specified in the Damage Tolerance Requirements per MIL-A-83444.

Stress-corrosion cracking may also occur in fatigue-critical components. This means that in addition to extension by fatigue, cracks might show some growth by stress corrosion. In dealing with this problem consider the following facts:

- Stress-corrosion cracking is a phenomenon that basically occurs under a steady stress. Hence, the in flight stationary stress level (1g) is the governing factor. Most fatigue cycles are of relatively short duration and do not contribute to stress-corrosion cracking. Moreover, the cyclic crack growth would be properly treated already on the basis of data for environment-assisted fatigue-crack growth. (If stress-corrosion cracking has to be accounted for, the stress-corrosion crack-growth rate should be superposed on the fatigue-crack-growth rate)
- Stress-corrosion cracking is generally confined to forgings, heavy extrusions, and other heavy sections, made of susceptible materials. Thus, the problem is generally limited to cases where plane strain prevails.
- The maximum crack size to be expected in service is $a_c = K_{Ic}^2/\pi\beta^2\sigma^2$, where σ equals σ_{LT} or σ_{DM} depending upon the inspectability level.

If stress-corrosion cracking is not accepted at any crack size, the 1-g stress, σ_{1g}, should be lower than $\sigma_{scc} = K_{Iscc}/\beta\sqrt{\pi a_c}$. With a_c given as above, it follows that complete prevention of stress corrosion extension of a fatigue crack requires selection of a material for which:

$$K_{Iscc} > \frac{\sigma_{1g}}{\sigma_{LT}\,(\text{or } \sigma_{DM})} K_{Ic} \quad . \tag{5.10}$$

5.2 VARIABLE-AMPLITUDE LOADING

5.2.1 Introduction

Baseline fatigue data are derived under constant-amplitude conditions. Crack-growth predictions have to be made for aircraft parts and components that are generally subjected to a stress history of variable amplitude. If there were no interaction effects of high and low cycles in the sequence, it would be relatively easy to establish a crack-growth curve by means of a cycle-by-cycle integration (see Section 5.2.5). However, crack growth under variable-amplitude cycling is largely complicated by interaction effects of high and low loads.

In the following sections these interaction effects will be briefly discussed. Crack-growth-prediction procedures taking interaction effects into account will be presented in Section 5.2.5.

5.2.2 Retardation

A HIGH LOAD OCCURING IN A SEQUENCE OF LOW-AMPLITUDE CYCLES SIGNIFICANTLY REDUCES THE RATE OF CRACK GROWTH DURING A LARGE NUMBER OF CYCLES SUBSEQUENT TO THE OVERLOAD. THIS PHENOMENON IS CALLED RETARDATION. Figure 5-16 shows a baseline crack-growth curve obtained in a constant-amplitude test (34). In a second experiment the same constant-amplitude loading was interspersed with overload cycles. After each application of the overload, the crack virtually did not grow during many cycles, after which the original crack-growth behavior was gradually restored.

Retardation is a result of residual compressive stresses at the crack tip. At the overload a large plastic zone is formed. The material in this zone undergoes a permanent stretch. Upon load release the surrounding material is elastically unloaded. It retracts to its original size in which the plastic zone material does not fit anymore due to its permanent elongation. In order to make it fit, the surrounding elastic material squeezes the plastic zone, which results in residual compressive stresses at the crack tip. These residual stresses are superposed on the subsequent cyclic stresses. Therefore the cyclic stresses are less effective in producing crack growth. When the crack tip has gradually grown through the region with compressive stresses, it resumes its original growth pattern.

In addition to the residual stresses in front of the crack tip, retardation may be affected by crack closure (35). At any one time there is a small plastic zone at the tip of the fatigue crack. When the crack has grown through the plastic zone, it leaves plastically deformed material in its wake. This material is permanently stretched. As a consequence, the crack faces will meet upon unloading before the load reaches zero (crack closure). Hence, there exists a system of residual compressive stresses in the wake of the crack also.

If the tensile overload is followed by a compressive overload, the material at the crack tip undergoes reverse plastic deformation. This reduces the residual stresses. Thus, a negative overload in whole or in part annihilates the beneficial effect of tensile overloads, as is also shown in Figure 5-16.

Retardation depends upon the ratio between the magnitude of the overload and subsequent cycles. This is illustrated in Figure 5-17. Sufficiently large overloads may cause total crack arrest. Hold periods at zero stress can partly alleviate residual stresses and thus reduce the retardation effect, (36,37) while hold periods at load increase retardation. Multiple overloads significantly enhance the retardation. This is shown in Figure 5-18.

5.2.3 Retardation Under-Spectrum Loading

In an actual service load history high- and low-stress amplitudes, and positive and negative "overloads" occur in random order. Retardation and annihilation of retardation becomes complex, but qualitatively the behavior is similar to that in a constant-amplitude history with incidental overloads. THE HIGHER THE MAXIMUM STRESSES IN THE SERVICE LOAD HISTORY, THE LARGER THE RETARDATION EFFECT DURING THE LOW-AMPLITUDE CYCLES. NEGATIVE STRESS EXCURSIONS DECREASE RETARDATION, AND TEND TO ENHANCE CRACK GROWTH.

These effects have been observed repeatedly (e.g., 38-44). They can best be illustrated by means of the data (38,39) shown in Figures 5-19 and 5-20. In random flight-by-flight simulation tests the highest load excursions were clipped to lower and lower levels (i.e., the magnitude of the high loads was reduced but no cycles were omitted). Figure 5-19 shows three crack-growth curves for three clipping levels. Lower clipping levels result in shorter crack-growth lives. Negative stress excursions reduce the retardation effect. Omission of the ground-air-ground cycles (negative loads) in the tests with the highest clipping level resulted in a larger crack-growth life for the same amount of crack growth.

Figure 5-20 shows the importance of load sequence. The crack-propagation life for random load cycling is shown at the top. Ordering the sequences of the loads, lo-hi, lo-hi-lo, or hi-lo increases the crack-growth life, the more so if the block size is larger. Hence, ordering is only permitted if the block size is small. Lo-hi ordering gives more conservative results than hi-lo ordering. In the latter case the retardation effect caused by the highest load is effective during all subsequent cycles.

5.2.4 Retardation Models

Some mathematical models have been developed to account for retardation in crack-growth-integration procedures. All models are based on simple assumptions, but within certain limitations and when used with experience each of them can produce results that can be used with reasonable confidence. The two yield zone models by Wheeler (45) and by Willenborg, et al (46), and the crack-closure model by Bell and Creager (47) will be briefly discussed. Detailed information and applications can be found in References 47-50.

Wheeler defines a crack-growth reduction factor, C_p:

$$\left(\frac{da}{dN}\right)_r = C_p \; f(\Delta K) \quad , \tag{5.11}$$

where f(K) is the usual crack-growth function, and (da/dN) is the retarded crack-growth rate. The retardation factor, C_p, is given as

$$C_p = \left(\frac{r_{pi}}{a_o + r_{po} - a_i}\right)^m \quad , \tag{5.12}$$

in which (see also Figure 5-21):

r_{pi} is the current plastic zone size in the ith cycle under consideration
a_i is the current crack size
r_{po} is the plastic size generated by a previous higher load excursion
a_o is the crack size at which the higher load excursion occurred
m is an experimental constant.

There is retardation as long as the current plastic zone is contained within a previously generated plastic zone.

Some examples of crack-growth predictions made by means of the Wheeler model are shown in Figure 5-22. Selection of the proper value for the exponent m will yield adequate crack-growth predictions. THE EXPONENT m IS SPECTRUM DEPENDENT AND ITS INDISCRIMINANT USE WITH RADICALLY DIFFERENT SPECTRA THAN FOR WHICH IT WAS DERIVED CAN LEAD TO INACCURATE AND UNCONSERVATIVE RESULTS.

The Willenborg model also relates the retardation to the overload plastic zone. It makes use of an effective stress-intensity factor, the maximum stress intensity in the ith cycle, K_{maxi}, being reduced to $K_{max,eff}$ as:

$$K_{max,eff} = K_{max,i} - \phi\left\{K_{max,o}\sqrt{1 - \frac{a_i - a_o}{r_{po}}} - K_{max,i}\right\} ,$$

$$\text{also:}\quad K_{min,eff} = K_{min,i} - \phi\left\{K_{max,o}\sqrt{1 - \frac{a_i - a_o}{r_{po}}} - K_{max,i}\right\} , \tag{5.13}$$

in which (see also Figure 5-21):

- ϕ is unity for the original model
- a_i is the current crack size
- a_o is the crack size at the occurrence of the overload
- r_{po} is the yield zone produced by the overload
- $K_{max,o}$ is the maximum stress intensity of the overload.

Since K_{max} and K_{min} are reduced by the same amount, the overload causes only a reduction in R ratio as long as $K_{min,eff} > 0$. When $K_{min,eff} < 0$, it should be taken as zero. In that case $R = 0$ and $\Delta K = K_{max,eff}$.

The equations show that retardation will occur until the crack has reached the boundary of the overload yield zone. At that time $a_i - a_o = r_{po}$ such that the reduction becomes zero.

Immediately after the overload $a_i = a_o$, which means that $K_{max,eff} = 2 K_{max,i} - K_{max,o}$. Consequently, the model predicts complete retardation ($K_{max,eff} = 0$) for the case the overload is twice as high as the subsequent cycle. Therefore, Gallagher and Hughes(51) introduced the factor ϕ in Equations (5.13), given by

$$\phi = \frac{K_{max,i} - K_{max,th}}{K_{max,o} - K_{max,i}} , \tag{5.14}$$

where $K_{max,th}$ is the threshold value (Section 5.1.3). Results of predictions made by means of the Willenborg model are presented in Figure 5-23.

Shortcomings to both the Wheeler model and the Willenborg model are:

— THE MODELS DO NOT ACCOUNT FOR THE REDUCTION OF RETARDATION BY NEGATIVE LOADS

— THE MODELS DO NOT DISTINGUISH BETWEEN A SINGLE OVERLOAD AND MULTIPLE OVERLOADS.

The crack-closure model by Bell and Creager attempts to overcome these shortcomings. This model makes use of a crack-growth-rate equation based on an effective stress-intensity range ΔK_{eff}. The effective stress intensity is the difference between the applied stress intensity and the stress intensity for crack closure. The latter is determined semiempirically in a very artificial way. The final equations contain many experimental constants, which reduces the versatility of the model. Details can be found in Reference 42. Some examples of predictions made with the model are presented in Figure 5-24 Because of the large number of empirical constants it is difficult to apply.

Crack-growth calculations are the most useful for comparative studies, where variations of only a few parameters are considered (i.e., trade-off studies to determine design details, design stress levels, material selection, etc.). THE PREDICTIONS SHOULD ALWAYS BE CHECKED WITH A FEW EXPERIMENTS. (See Analysis Substantiation Tests in Section 7.1.) Then other predictions o a similar nature can be used with greater confidence. After a discussion of spectrum and stress-history development, example calculations of crack-growth curves will be given in Section 5.4.

The shortcomings of the retardation models are not the only cause of uncertainties in crack-growth predictions. Other factors contributing to the uncertainty are:

-- Scatter in baseline da/dn data

-- Unknowns in the effects of service environment

-- Necessary assumptions on flaw shape development (Section 5.4.4)

-- Deficiencies in K calculation (sec 5.4.4)

-- Assumptions on interaction of cracks (see 5.4.5)

-- Assumptions on service stress history (see 5.3).

In view of these problems (to be discussed in later sections), the use of more sophisticated retardation models would not greatly improve the reliability of crack-growth predictions of service behavior.

In view of these additional shortcomings of crack-growth predictions, the shortcomings of a retardation model become less pronounced; therefore, no particular retardation model has preference over the others. From a practical point of view, the Willenborg model is easier to use since it does not contain empirical constants.

5.2.5 Computer Routines

Several computer programs are available for general use that include one or more of the retardation models in a crack-growth-integration scheme. The most well known of these is CRACKS* (50), the latest version of which should be used. It has the options of using any of the three retardation models discussed in the previous section. However, most companies have their own computer programs.

In general, the crack-growth-damage-integration procedure consists of the following basic steps (Figure 5-25).

Step 1. The initial crack size follows from the damage tolerance assumption as a_1. The stress range in the first cycle is $\Delta\sigma_1$. (see Section 5.4) Then determine $\Delta K_1 \approx \beta\Delta\sigma_1\sqrt{\pi a_1}$ by using the appropriate β for the given structural geometry and crack geometry. (The computer program may include the determination of β,)

Step 2. Determine $(da/dN)_1$ at ΔK_1 from the da/dN - ΔK baseline information, taking into account the appropriate R value. (The da/dN - ΔK baseline information may make use of one of the crack-growth equations discussed in Section 5.1.3. The computer program may contain options for any one of these equations. Instead it may use data in tabular form and interpolate between data points.)

The crack extension Δa_1 in cycle 1 is

$$\Delta a_1 = \left(\frac{da}{dN}\right)_1 \times 1.$$

* Available through AFFDL/FBEC, Fatigue, Fracture and Reliability Group.

The new crack size will be $a_2 = a_1 + \Delta a_1$.

Step 3. The extent of the yield zone in Cycle 1 is determined as

$$Y_2 = a_o + r_{pl} \text{ and call } a_o = a_1$$

with $r_{pl} = \frac{K_{max}^2}{2\pi\sigma_{ys}^2}$ for plane stress

or $r_{pl} = \frac{K_{max}^2}{4\sqrt{2}\pi\sigma_{ys}^2}$ for plane strain.

Step 4. The crack size is now a_2. The stress range in the next cycle is $\Delta\sigma_2$. Calculate ΔK with $\Delta K_2 = \beta\Delta\sigma_2\sqrt{\pi a_2}$.

Step 5. Calculate the extent of the yield zone

$$Y_{22} = a_2 + r_{p2}.$$

Step 6. If $Y_{22} < Y_2$ calculate C_p according to Equation (5.12) when using the Wheeler model, or calculate $K_{max,eff}$, $K_{min,eff}$ and R according to Equations (5.13) when using the Willenborg model. Skip Steps 7 and 8, go to Step 9.

Step 7. If $Y_{22} \geq Y_2$, determine $(da/dN)_2$ from ΔK_2. Determine the new crack size

$$a_3 = a_2 + \Delta a_2 = a_2 + \left(\frac{da}{dN}\right)_2 \times 1.$$

Step 8. Replace Y_2 by Y_{22} which is now called Y_2.
Replace $a_o = a_1$ by $a_o = a_2$.
Skip Step 9, go to Step 10.

Step 9. When using the Wheeler model, determine the amount of crack growth on the basis of ΔK_2 from the da/dN – ΔK data. Find the new crack size from

$$a_3 = a_2 + \Delta a_2 + C_p\left(\frac{da}{dN}\right)_2 \times 1.$$

When using the Willenborg model, determine the amount of crack growth using the ΔK_{eff} and R value determined in Step 6 from the da/dN – ΔK plot. Determine the new crack size as

$$a_3 = a_2 + \Delta a_2 = a_2 = (\frac{da}{dN})_{2\ eff} \times 1.$$

Step 10. Repeat Steps 4 through 9 for every following cycle, while for the ith cycle replacing a_2 by a_i and a_3 by a_{i+1}.

This routine of cycle-by-cycle integration is not always necessary. The integration is faster if the crack size is increased stepwise in the following way.

- At a certain crack size the available information is a_i, a_o, Y_2.
- Calculate Δa_i for the i_{th} cycle in the same way as in Steps 4 through 9.
- Calculate Δa_j ..... Δa_n for the following cycles but let the current crack size remain a_i all the time. This eliminates recalculation of β every cycle.
- Calculate Y_{2k} for every cycle. If $Y_{2k} > Y_2$, then replace Y_2 by Y_{2k} and call it Y_2. Then replace a_o by a_i and call it a_o.
- Sum the crack-growth increments to

$$\delta a = \sum_{k=i}^{n} \Delta a_k \quad .$$

- Continue until Δa exceeds a certain preset size. Then increment the crack size by

$$a = a_i + \Delta a,$$

and repeat the procedure.

A reasonable size for the crack-growth increment is $\Delta a = \frac{1}{20} a_i$. It can also be based on the extent of the yield zone, e.g., $\Delta a = \frac{1}{10} (Y_2 - a_i)$. The advantage of the incremental crack-growth procedure is especially obvious if series of constant-amplitude cycles occur. Since the crack size does not change, the stress intensity will not change. Hence, each cycle will cause the same amount of growth. This means that all n constant-amplitude cycles can be treated as one cycle to give

$$\Delta a = n \frac{da}{dN} \quad .$$

There exist other possibilities for more efficient integration schemes. However, their use is largely determined by the type of stress history that has to be integrated. For example, for a flight-by-flight load history it is possible to determine the crack extension per flight as

$$\frac{da}{dF} = \sum_{i=1}^{n} \left(\frac{da}{dN}\right)_i \quad , \tag{5.15}$$

if there are n cycles in the flight. This is done for a number of crack sizes and for all the different flight types (e.g., missions) in the sequence. Then a diagram is constructed of da/dF versus a for each flight type, which can be integrated on a flight-by-flight basis or on a crack-increment basis.

THE INTEGRATION SCHEME IS A MATTER OF PERSONAL TASTE, AND OF AVAILABLE FACILITIES. This means that there is no preference for the use of a particular computer program other than those dictated by computer facilities.

5.3 STRESS SPECTRA

5.3.1 Exceedance Spectra and Their Use

In order to predict the crack-growth behavior of an aircraft structure, the designer needs to know the stress history. For a new design the stress history can onl e estimated on the basis of measurements made on existing aircraft systems. As pointed out in the previous sections, the stress history, or more specifically the sequence of low and high stresses is of great influence on the predicted crack-growth life. Not only the magnitude of the stresses but also the sequence in which they occur is of importance.

Measurements made on existing airplanes do give fairly accurate information as to the magnitude of the aircraft loads, but the information on the sequence of loads is only rudimentary. Nevertheless, assumptions have to be made regarding the stress history in order to enable crack-growth predictions. This section discusses this problem by pointing out the difficulties involved, without giving rules to arrive at a stress history because that is beyond the scope of this document.

The load information for an aircraft structure is usually in the form of an exceedance spectrum. The spectrum is already an interpretation of in-flight measurements. The flight measurements either pertain to center of gravity accelerations or stresses at a particular location. The interpretation consists of a counting procedure, which counts accelerations (or stre of a certain magnitude, or their variation (range). Information on the various counting procedures can be found in References 52 and 53.

Typical exceedance spectra are given in Figure 5-26 for a transport wing, bomber wing, and fighter wing. The ordinate either can be accelerations or stresses (in some cases gust velocities). The abscissa represents the number of times a level on the vertical axis is exceeded. E.g., in Figure 5-26a level A is exceeded n_1 times; level B is exceeded n_2 times. This means that there will be n_1-n_2 events of a load between levels A and B. These loads will be lower than B, but higher than A. The exact magnitude of any one of the n_1-n_2 loads remains undetermined.

Basically, one can define an infinite number of load levels between A and B. However, there are only n_1-n_2 occurrences, which means that the number of load levels to be encountered is finite; not every arbitrary load level will be experienced. Strictly speaking each of the n_1-n_2 occurences between A and B could be a different load level. If one chose to divide the distance between A and B into n_1-n_2 equal parts, ΔA, each of these could occur once. Mathematically, a level A+ΔA will be exceeded n_1-1 times. Hence, there must be one occurrence between A and A+ΔA. In practice such small steps cannot be defined, nor is there a necessity for their definition.

If measurements were made again during an equal number of flight hours, the exceedance spectrum would be the same, but the actual load containment would be different. This means that the conversion of a spectrum into a stress history for crack-growth analysis will have to be arbitrary because one can only select one case out of unlimited possibilities.

Going to the top of the spectrum in Figure 5-26, level C will be exceeded 10 times. There must be a level above C that is exceeded 9 times, one that is exceeded 8 times, etc. One could identify these levels, each of which would occur once. In view of the foregoing discussion this becomes extremely unrealistic. Imagine 10 levels above C at an equal spacing of ΔC, giving levels C, C+ΔC, C+2ΔC, etc. If level C is exceeded 10 times, all of these exceedances may be of the level C+3ΔC for one aircraft; they may be all of level C+5ΔC for another aircraft.

As a consequence, it is unrealistic to apply only one load of a certain level, which would imply that all loads in the history would have a different magnitude. Moreover, if high loads are beneficial for crack growth (retardation), it would be unconservative to apply once the level C+ΔC, once C+2ΔC, etc., if some aircraft would only see 10 times C.

Hence, the maximum load level for a fatigue analysis should be selected at a reasonable number of exceedances. (This load level is called the clipping level.) From crack-growth experiments regarding the spectrum clipping level, it appears resonable to select the highest level at 10 exceedances per 1,000 flights. This will be discussed in more detail in later sections. (NOTE THAT THE MAXIMUM LOAD USED IN THE FATIGUE ANALYSIS HAS NO RELATION WHATSOEVER TO THE P_{xx} LOADS FOR RESIDUAL STRENGTH ANALYSIS.)

The same dilemma exists when lower load levels have to be selected. Obviously, the n loads in 1,000 hours will not be at n different levels. A number of discrete levels has to be selected. This requires a stepwise approximation of the spectrum, as in Figure 5-27. As shown in the following table, the number of occurrences of each level follows easily from subtracting exceedances.

Level	Exceedances	Occurrences
L_1	n_1	n_1
L_2	n_2	$n_2 - n_1$
L_3	n_3	$n_3 - n_2$
L_4	n_4	$n_4 - n_3$
L_5	n_5	$n_5 - n_4$

The more discrete load levels there are, the closer stepwise approximation will approach the spectrum shape. On the other hand, the foregoing discussion shows that too many levels are unrealistic. The number of levels has to be chosen to give reliable crack-growth predictions.

Figure 5-28 shows results of crack-growth calculations in which the spectrum was approximated in different ways by selecting a different number of levels each time. If the stepped approximation is made too coarse (small number of levels) the resulting crack growth curve differs largely from those obtained with finer approximations. However, if the number of levels is 8 or more, the crack-growth curves are identical for all practical purposes. A further refinement of the stepped approximation only increases the complexity of the calculation; it does not lead to a different (or better) crack-growth prediction. Crack-growth predictions contain many uncertainties anyway, which means that one would sacrifice efficiency to apparent sophistication by taking too many levels. IT TURNS OUT THAT 8 TO 10 POSITIVE LEVELS (ABOVE THE IN-FLIGHT STATIONARY LOAD) ARE SUFFICIENT. THE NUMBER OF NEGATIVE LEVELS (BELOW THE IN-FLIGHT STATIONARY LOAD) MAY BE BETWEEN 4 AND 10.

Selection of the lowest positive level is also of importance, because it determines the total number of cycles in the crack-growth analysis. (This level is called the truncation level.) Within reasonable limits the lower truncation level has only a minor effect on the outcome of the crack-growth life. THEREFORE, IT IS RECOMMENDED THAT THIS LOWER TRUNCATION LEVEL BE SELECTED ON THE BASIS OF EXCFEDANCES RATHER THAN ON STRESSES. A NUMBER IN THE RANGE OF $10^5 - 5 \times 10^5$ EXCEEDANCES PER 1,000 FLIGHTS SEEMS REASONABLE. This will be discussed in more detail in later sections.

As an example, consider the spectrum in Figure 5-27. (Only the positive part is shown.) The highest level, L_1, is selected at 10 exceedances; the lowest level, L_5, at 10^5 exceedances. The intermediate levels are evenly spaced between L_5 and L_1, giving the following results:

Level	Exceedances	Occurrences
L_1	10	10
L_2	1,000	990
L_3	7,000	6,000
L_4	25,000	18,000
L_5	100,000	75,000
	TOTAL	100,000

The stepped approximation is constructed as follows. The maximum level, L_1, is selected at e.g., the load that is exceeded 10 times in 1,000 flights. The other levels are selected between 1g and L_1. They need not necessarily be at equal spacing. Then the steps are constructed by ensuring that (Figure 5-27) the hatched area A_2 is equal to B_2, $A_3 = B_3$ and so on. At the highest level step is constructed as indicated in Figure 5-27, otherwise too many high loads will occur.

5.3.2 Design Spectra

Requirements for the development of aircraft load histories are presented in MIL-A-8866B (USAF). The individual spectra of repeated loads have to be assembled on a flight-by-flight basis. The sources of repeated loads shall include run-ups, check-outs, jacking, towing, taxiing, landing, maneuvering, turbulence, inflight-refueling, control operations, pressurization, buffeting. In other words, all possible sources of cyclic loads shall be included.

In order to derive the load history, a flight has to be divided into segments as in Figure 5-29. A load sequence has to be determined for each mission on the basis of available spectrum information. MIL-A-8866B (USAF) presents typical maneuver spectrum information for different missions, mission segments, and different aircraft systems. The number of (positive) levels given is on the order of 10. The total number of exceedances is on the order of 10^5 - 5.10^5. These typical data can be converted into exceedance spectra. The result is shown in Figure 5-30 on a 1,000 flight-hours basis.

The spectrum information has to be converted into stresses at the critical locations. The exceedance curve of accelerations is the integral of all loads the aircraft experiences during a certain period of operations. During this time the airplane will have different weights and configurations (e.g., external stores, flaps) and it will fly a variety of missions under a variety of weather conditions. THIS MEANS THAT A GIVEN ACCELERATION IN THE SPECTRUM IS NOT RELATED TO ONE UNIQUE STRESS LEVEL AT THE CRITICAL LOCATION. RATHER, THERE EXISTS A COMPLICATED CORRELATION FUNCTION BETWEEN THE ACCELERATION AND THE INDUCED STRESS.

Before the conversion into stresses, the total spectrum is divided into mission-spectra, and where necessary, into spectra for the various mission segments. Such a division is also made in MIL-A-008866B (USAF). First, the different missions are defined. Then the mission profiles are established. Next the spectrum for each mission (and mission segment) is derived from existing data or the above specification. Using the correlation function, aircraft weight, and aircraft configuration, the mission stresses can be determined.

5.3.3 Flight Stress History; Mission Mix

After the determination of mission stresses, a flight stress history and a mission mix have to be determined. The sequencing of stresses and missions may significantly affect the outcome of an experiment or a crack-growth calculation, due to retardation effects. Some guidelines for sequencing are given below.

(a) Deterministic loads are placed in the proper flight segment. Obviously, the ground load of the ground-air-ground cycle will occur at the beginning and at the end of each flight. Similarly, maneuver loads associated with take-off will be at the beginning of the flight. Specific maneuver loads (e.g., those associated with strikes) should be positioned in the proper mission segment.

(b) Probabilistic loads due to gusts and maneuvers have to be arbitrarily sequenced. A random sequence would be the most realistic. Taking a random sequence for mission A, and making all flights of mission A equal would largely devaluate the random character. A different order in each flight of mission A reduces the efficiency of the integration procedure. Hence, one would want to decide for a standard mission A. Then it is also reasonable to adopt a cycle sequence that is relatively simple without sacrificing the accuracy of crack-growth predictions. Test results (38,39) indicate that low-high-low sequencing within a flight for a gust spectrum gives crack-growth lives very similar to pure random loading. For a fighter spectrum lo-hi-lo loading per flight is realistic because most of the maneuvering will take place during the combat phase about halfway through the flight. THEREFORE, LO-HI-LO SEQUENCING PER FLIGHT IS RECOMMENDED IF PROGRAMMED SEQUENCING IS CONSIDERED INSTEAD OF RANDOM SEQUENCING. If other than random or lo-hi-lo sequencing per flight is selected, the adequacy of this choice should be demonstrated by analysis and tests.

(c) If the spectrum used represents about 1,000 flight hours (as in MIL-8866B (USAF), it will be a spectrum for a rather limited number of flights. For example, if the average flight duration is 5 hours, the spectrum will be for 200 flights. This means that any stress level that is exceeded less than 200 times will not occur in every flight. If the clipping level is selected at 10 exceedings per 1,000 flights, it will occur only once very 100 flights. This problem is easily resolved if the highest levels occur only in one mission type, and if that mission occurs once or twice in 200 flights. In that case each stress occurrence in the spectrum will be automatically accounted for (see Section 5.3.4). Otherwise, the standard missions will have to be modified to account for the high stresses. For example, in the above example, every 100 flights one mission is modified to contain the highest level.

(d) A realistic mission sequence (mission mix) has to be established.

If the number of mission types is relatively small, and the number of flights of each mission is realtively large, ordering is not very critical. However, usually there is a small number of severe missions in which the highest stresses occur. If they all occur at the beginning of the life, the retardation effect may be excessive. If they all occur at the end of the life, the retardation effect is absent, which would give a conservative crack-growth prediction. However, conservatism is already built in by means of the clipping level. THEREFORE, WITH NO OTHER INFORMATION AVAILABLE, IT IS RECOMMENDED THAT THE SEVEREST MISSIONS BE EVENLY SPACED AMONG THE OTHER MISSIONS.

(e) After determination of all the mission stresses, simplifications are sometimes possible. Usually the stresses will be given in tabular form. They will show an apparent variability. For example, if an acceleration, n_2, is exceeded 10,000 times, this will not result in the exceedance of 10,000 times of a certain stress level, since n_2 causes a different stress in different missions or mission segments. However, if a stress exceedance spectrum is established of the various missions on the basis of the tabular stress history, it may turn out that two different missions may have nearly the same stress spectrum. In that case, the missions can be made equal for the purpose of crack-growth predictions.

5.3.4 Simple Spectra and Simpiification of Spectra

So far, the spectra considered were applicable primarily for airplane wings. Fin and stabilator experience a combined gust and maneuver spectrum that is usually as complicated as that of the wing. However, both structures operate essentially at zero mean load, which means that ground-air-ground cycles are of little significance. The derivation of a stress history for these parts follows the same rules as for the wing.

Parts with relatively simple spectra are the flap structure components. These experience one cycle during take off, and one cycle during landing. Maneuvering and gust cycles are superposed to them, but they are so small as compared to the primary loading, that they can usually be neglected.

Fuselage structures are subjected to torsional and bending loads due to maneuvering and gust loads on the control surfaces, aerodynamic loads, and pressurization cycles. The latter will be the only significant loading for many locations in the fuselage.

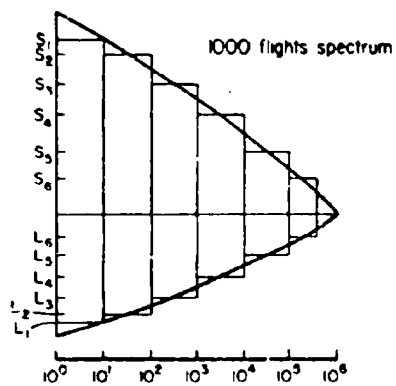
In the early design stage not much is known about stress histories to be anticipated. An exceedance spectrum based on previous experience is usually available. However, material selection may still have to be made, and operational stress levels may still have to be selected. Hence, it is impossible and premature to derive a service stress history. Yet, crack-growth calculations have to be made as part of the design trade-off studies. The designer wants to know the effect of design stress, structural geometry, and material selection with respect to possible compliance with the damage-tolerance criteria, and with respect to aircraft weight and cost. Such studies can be made only if a reasonable service stress history is assumed. The following procedure shows how such a history can be derived in a simple way, if it is to be used only for comparative calculations.

Consider the exceedance spectrum for 1,000 flights shown in Table 5.1. Instead of selecting stress levels for the discretization it is much more efficient in this case to select exceedances. Since a large number of levels is not necessary in this stage, six levels were chosen in the example. The procedure would remain the same if more levels were to be selected.

The exceedances in the example were taken at 10 (in accordance with Section 5.3.1); 100; 1,000; 10,000; 100,000; 500,000 (in accordance with Section 5.3.1). Vertical lines are drawn at these numbers, and the stepped approximation is made. This leads to the positive excursion levels, S_1 - S_6, and the negative excursion levels, T_1 - T_6 (Table 5-1). The stress levels and exceedances are given in columns 1 and 2 of Table 5-1. Subtraction gives the number of occurrences in column 3.

The highest stress level is likely to occur only once in the severest mission. Therefore, a mission A spectrum is selected as in column 4 in which S_1 occurs once, and lower levels occur more frequently in accordance with the shape of the total spectrum. In order to use all 10 occurrences of level S_1, it is necessary to have 10 missions A in 1,000 flights. These

TABLE 5.1 SIMPLE FLIGHT-BY-FLIGHT SPECTRUM FOR EARLY DESIGN ANALYSIS AND TRADE-OFF STUDIES



MISSION MIX

33 FLIGHTS BLOCK:
- 6 TIMES MISSION D
- 1 TIME MISSION B
- 19 TIMES MISSION C
- 1 TIME MISSION B
- 6 TIMES MISSION D

REPEAT BLOCK; EVERY THIRD BLOCK APPLY 1 TIME MISSION A.

COMBINE: S_1 WITH L_1 TO ONE CYCLE
S_2 WITH L_2 TO ONE CYCLE
ETC.

SEQUENCE: LO-HI-LO IN FLIGHT.

			Mission A			Mission B			Mission C			Mission D		
① Level	② Exceedings	③ Occurrences	④ Occurrences	⑤ 10X	⑥ Remainder 3 - 5	⑦ Occurrences	⑧ 60X	⑨ Remainder 6 - 8	⑩ Occurrences	⑪ 57X	⑫ Remainder 9 - 11	⑬ Occurrences	⑭ 360X	⑮ Remainder 12 - 14
S_1	10	10	1	10	--	--	--	--	--	--	--	--	--	--
S_2	100	90	3	30	60	1	60	--	--	--	--	--	--	--
S_3	1,000	900	15	150	750	3	180	570	1	570	--	--	--	--
S_4	10,000	9,000	48	480	8,520	17	1,020	7,500	10	5,700	1,800	5	1,800	--
S_5	100,000	90,000	300	3,000	87,000	200	12,000	75,000	100	57,000	18,000	50	18,000	--
S_6	500,000	400,000	1,900	19,000	381,000	1,500	90,000	291,000	400	228,000	63,000	175	63,000	--

10 missions A will use the numbers of cycles given in column 5. When the 10 missions A are subtracted from the total number of occurrences in column 3, the remaining load containment of the remaining 990 flights is as given in column 6.

The next severest mission is likely to have one cycle of level S_2. Hence, the mission B spectrum in column 7 can be constructed in the same way as the mission A spectrum. There remain 60 cycles of S_2. Hence, mission B will occur 60 times in 1,000 flights. The 60 missions B will use the cycles shown in column 8. Therefore, the cycles remaining for the remaining 930 flights are as given in column 9.

Level S_3 will occur once in a mission C, which is constructed in column 10. There remain 570 cycles S_3, so that there will be 570 missions C. These will use the cycles given in column 11. The remaining cycles are given in column 12.

There will be 10 missions A, 60 missions B, and 570 missions C in 1,000 flights. This means that there remain 360 flights. Dividing the remaining cycles in column 12 by 360, a mission D spectrum is found as in column 13. Consequently, all cycles have been accounted for.

A mission mix has to be constructed now. With mission A occurring 10 times per 1,000 flights, a 100-mission block could be selected. However, a smaller block would be more efficient. In the example, a 33-mission block can be conceived as shown in Table 5-1. After 3 repetitions of this block (99 flights) one mission A is applied.

The cycles in each mission are ordered in a lo-hi-lo sequence. The negative excursion T_1 - T_6 are accounted for by combining them with the positive excursions of the same frequency of occurrence: T_1 forms a cycle with S_1, T_2 with S_2, etc. In this way the range of a cycle is S-T, instead of S-mean stress, which is conservative.

In order to arrive at the stresses an approximate procedure has to be followed also. Given the flight duration, an acceleration spectrum (e.g., the 1,000 hours spectra given in MIL-A-8866B) can be converted approximately into a 1,000 flight spectrum. Limit load will usually be at a known value of n_z (e.g., 7.33g for a fighter or 2.5 g for a transport). As a result, the vertical axis of the acceleration diagram can be converted into a scale that gives exceedances as a fraction of limit load. This is done in Figure 5-31 for the MIL-A-8866B spectra of Figure 5-30. A comparison of these figures will clarify the procedure.

Once a spectrum of the type of Figure 5-31 is established, design trade-off studies are easy. Selecting different materials or different design stress levels $S_1 - S_6$ and $T_1 - T_6$ can be determined and the flight-by-flight spectrum is ready (Table 5-1). Selection of a different design stress level results in a new set $S_1 - S_6$. This requires only the exchange of a few cards in the computer program, and the calculation can be rerun.

This shows the versatility of the spectrum derivation of Table 5-1. It is a result of choosing exceedances to arrive at the stepped approximation of the spectrum, which means that the cycle content is always the same. If stress levels were selected instead, a change in spectrum shape or stress levels would always result in different cycle numbers. In that case, the whole procedure to arrive at the spectrum of Table 5-1 would have to be repeated, and many more changes would have to be made to the computer program.

Of course, Table 5-1 is an example only. The spectrum could be approximated by more levels, more different missions could be designed, but the same procedure could still be used. In view of the comparative nature of the calculations in the early design stage, many more levels or missions are not really necessary.

NOTE: THE STRESS HISTORY DERIVED IN THIS SECTION IS USEFUL ONLY FOR QUICK COMPARATIVE CALCULATIONS FOR TRADE-OFF STUDIES.

The stress history developed in Table 5-1 was applied to all the spectra in Figure 5-31 to derive crack-growth curves. The results will be discussed in Section 5.4.3.

5.4 CRACK-GROWTH PREDICTION

5.4.1 Introduction

The analysis procedure for crack-growth prediction requires the following steps:

(1) Determine the stress-intensity factor (Chpater 4).

(2) Establish a stress history and mission mix (Section 5.3).

(3) Find baseline crack-growth data (Section 5.1).

(4) Select a retardation model (Section 5.2); select and apply an integration routine (Section 5.2).

Each of these steps was discussed in general terms in one of the foregoing sections. However, there are some detail problems that need consideration. These detail problems will be the subject of Section 5.4.

5.4.2 Cycle Definition and Sequencing

In Section 5.2, the retardation phenomenon was discussed. Retardation causes high stress excursions to have a large effect on crack growth. As a result, the sequence of low and high stresses can be very critical. However, there is another sequence effect that is not at all related to retardation. It is related to the cycle definition necessary for a crack-growth calculation.

If a flight-by-flight stress history is developed for damage tolerance analysis or tests, it will be given as a sequence of load levels. Each of the cases, a, b, c, and d in Figure 5-32, could be a detail of such a sequence. Each case is a stress excursion of 8δ between levels A and B containing a dip of increasing size from a to d. In case a, the dip is so small that it can be neglected. The cycle can be considered a single excursion with a range ΔK_1 of size 8δ. In case b, the dip cannot be neglected. A normal crack-growth calculation would consider case b a sequence of two excursions, one with a range ΔK_2, the other with a range ΔK_3, each of size 5δ.

If the four cases were treated this way, the normally calculated crack extension would be as given in the center of Figure 5-32. (For simplicity, the crack-growth equation is taken as $da/dN = C(\Delta K)^4$ and the R ratio effect is ignored). It turns out that cases b and c would cause considerably less crack extension than case a. This is very unlikely in practice, since the crack would see one excursion from A to B in each case. Therefore, cases b, c, and d should be more damaging than case a in view of the extra cycle due to the dip. Although the effect of cycle ratio was neglected, the small influence of R could not account for the discrepancies.

It seems more reasonable to treat each case as one excursion with a range of ΔK_1 plus one excursion of a smaller range (e.g., ΔK_4 in case b) (range-pair counting). If this is done, the ranges considered would be as indicated by the dashed lines in Figure 5-32. The alternative crack-growth calculation is shown also in Figure 5-32 (bottom). There is an increasing damage going from a to d.

The alternative cycle definition is obtained by a rainflow count (53,54). The method is illustrated in Figure 5-33. While placing the graphical display of the stress history vertical, it is considered as a stack of roofs. Rain is assumed to flow from each roof. If it runs off the roof, it drips down on the roof below, etc., with the exception that the rain does not continue on a roof that is already wet. The range of the rain flow is considered the range of the stress. The ranges so obtained are indicated by AB, CD, etc., in Figure 5-33. Figure 5-34 shows how rainflow counting may affect a crack-growth prediction.

Several other counting methods exist. They are reviewed in References 52 and 53. Counting methods were originally developed to count measured load histories in order to establish an exceedance diagram. Therefore, the opinions expressed in the literature on the usefulness of the various counting procedures should be considered in that light. The counting procedure giving the best representation of a spectrum need not necessarily be the best descriptor of fatigue behavior.

It is argued that ranges are more important to fatigue behavior than load peaks. On this basis, the so-called range-pair count (52,53) and the rainflow count (54) are considered the most suitable. However, no crack growth experiments were ever reported to prove this.

The use of counting procedures in crack-growth prediction is an entirely new application. An experimental program is required for a definitive evaluation. Calculated crack-growth curves show that the difference in crack-growth life may be on the order of 25-30 percent. It should be noted that counting is not as essential when the loads are sequenced lo-hi-lo in each flight.

The increasing ranges automatically produce an effect similar to counting.

FOR THE TIME BEING, IT SEEMS THAT A CYCLE COUNT WILL GIVE THE BEST REPRESENTATION OF FATIGUE BEHAVIOR. THEREFORE, IT IS RECOMMENDED THAT CYCLE COUNTING PER FLIGHT BE USED FOR CRACK-GROWTH PREDICTIONS OF RANDOM SEQUENCES.

In the computer program for crack-growth integration, each flight is first cycle counted. The ranges resulting from this count are used to determine the K values. Care should be taken that they are sequenced properly in order to avoid different interaction effects (note that K_{max} determines retardation and not K). As an example, consider again Figure 5-33. The proper sequence for integration is: CD, GH, KL, EF, AB, PQ, MN. In this way, the maximum stress intensity (at B) occurs at the proper time with respect to its retardation effect. This way the maximum stress intensity of cycle AB will cause retardation for cycles PQ and MN only.

5.4.3 Clipping

Apart from the sequencing problems addressed in the previous section, there is a sequence problem associated with retardation. In Section 5.3, it was pointed out that sequencing of deterministic loads should be done in accordance with service practice; probabilistic loads can be sequenced randomly, but a lo-hi-lo order per flight is acceptable. This can be concluded from data of the type presented in Figure 5-20.

The sequencing effect due to retardation is largely dependent on the ratio between the highest and lowest loads in the spectrum and their frequency of occurrence. As a result, it will depend upon spectrum shape. Compare, for example, the fighter spectrum with the transport spectrum in Figure 5-31. The relatively few high loads in the transport spectrum may cause a more significant retardation effect than the many high loads that their exact magnitude and sequence becomes of less importance.

The selection of the highest loads in the load history is critical to obtain a reliable crack-growth prediction. It was argued in Section 5.3 that it is not realistic to include loads that occur less frequently than about 10 times in 1,000 flights, because some aircraft in the fleet may not see these high loads. It means that the spectrum is clipped at 10 exceedings. No load cycles are omitted. Only those higher than the clipping level are reduced in magnitude to the clipping level. The effect of clipping on retardation and crack-growth life was illustrated in Figure 5-19.

The question remains whether proper selection of a realistic clipping level is as important for a crack-growth prediction as it is for an experiment. In this respect, it is important to know which retardation model is the most sensitive to clipping level. As pointed out above, this may also depend upon spectrum shape. This was checked by running crack-growth calculations for different clipping levels, different spectrum shapes, and with two retardation models.

Calculations were made for the six spectra shown in Figures 5-31, by using the flight-by-flight history developed in Table 5-1. The cycles in each flight were ordered in a lo-hi-lo sequence. Crack-growth curves for the full spectra are shown in Figure 5-35 for the Willenborg model and in Figure 5-36 for the Wheeler model. The crack configuration is indicated in the figures. A stress of 35 ksi at limit load was taken for all spectra.

Subsequently, four significantly different spectra (A, B, C, and E) were selected. Crack-growth curves were calculated using the clipping levels S_2, S_3, S_4, and S_5 in Table 5-1. The resulting crack-growth curves for one spectrum are presented in Figure 5-37. Also shown is a curve for a linear analysis (no retardation). The results for all spectra are compiled in Figure 5-38. Test data for gust spectrum truncation are also shown. Some characteristic numbers are tabulated.

The figures allow the following observations:

-- The two models give largely different crack-growth lives for all spectra, except C. The differences are not systematic. Since there are no test data for comparison, the correct answers are not known. However, by changing the retardation exponent, the Wheeler calculations could be adjusted to match the test data.

-- With one exception the two models essentially predict the same trend with respect to clipping levels. This shows that they both have equ[illegible] capability to treat retardation. Hence, the Wheeler mode[illegible] have greater versatility for different spectrum shapes prov[illegible]ne retardation exponent is adjusted.

-- The steep spectra (fighter, trainer) are somewhat less sensitive to clipping level. Apparently, the damage of the high cycles outweighs their retardation effect.

-- With extreme clipping the analysis attains more the character of a linear analysis.

-- Bringing the clipping level down from 10 exceedings per 1,000 flights (top data points in Figure 5-38) to 100 exceedings per 1,000 flights (second row of data points in Figure 5-38) reduces the life by only 15 percent or less for all spectra.

In addition, crack-growth calculations were made to repredict the gust spectrum test data shown in Figure 5-38. The results are presented in Figures 5-39 and 5-40. It turns out that the calculated results are very conservative. However, with one exception, they would all fall within the scatterband of Figure 5-23. The baseline data used were worst case upper-boundary data. This can easily account for a factor two in growth rates. If the growth rates were reduced by a factor of two, the calculations would be very close to the test data (dashed line in Figure 5-40).

One important thing has been disregarded so far. Of the retardation models, only the one by Bell and Creager accounts for compressive stresses. As shown in Figure 5-16, compressive stresses reduce retardation (compare curves B and C. Omission of the g.a.g. cycle in the experiments (39) of Figure 5-40 increased the life by almost 80 percent. Apart from the g.a.g. cycle there are other compressive stresses in the spectrum. All of these were ignored in the calculation with the Willenborg model.

The top clipping level in Figure 5-40 is at 5 exceedings per 1,000 flights, the second level is at 13 exceedings per 1,000 flights. From these results and Figure 5-38, it appears that an exceedance level of 10 times per 1,000 flights will combine reasonable conservatism with a realistically high clipping level. THIS SUPPORTS THE ARGUMENTS GIVEN PREVIOUSLY TO SELECT THE CLIPPING LEVEL AT 10 EXCEEDINGS PER 1,000 FLIGHTS FOR BOTH CALCULATIONS AND EXPERIMENTS. THE EFFECT OF CLIPPING LEVEL SHOULD BE CALCULATED FOR A SMALL NUMBER OF REPRESENTATIVE CASES TO SHOW THE DEGREE OF CONSERVATISM.

5.4.4 Truncation

Truncation of the lower load levels is important for the efficiency of crack-growth calculations. Truncation means that cycles below a certain magnitude are simply omitted. The argument is that low stress excursions do not contribute much to crack growth, especially in view of the retardation effect. Since there are so many cycles of low amplitude, their omission would speed up experiments and crack growth calculations.

Figure 5-41 shows some experimental data regarding the effect of truncation. However, these data are somewhat misleading, because truncation was not carried out properly. The lowest load levels of a complete stress history were simply omitted, without a correction of the stress history. Figure 5-42 shows the improper and the correct procedure for truncation.

The left half of Figure 5-42 illustrates the truncation procedure used for the experiments in Figure 5-41. In the example, the 580,000 cycles of level S_8 would simply be omitted, thus reducing the total cycle content from 7000,000 to 120,000. Proper truncation requires that the lower spectrum approximation step be reconstructed, as in the right half of Figure 5-42. The hatched areas in the figure should be made equal. This means that the number of S_7 cycles would increase from 80,000 to 260,000, the total cycle content would be reduced from 700,000 to 300,000. In this way, 180,000 cycles S_7 would be substituted for 580,000 cycles S_8. In this way, the effects of lower level truncation are less than suggested by the experimental data in Figure 5-41.

IN SECTION 5.3 IT WAS RECOMMENDED THAT THE TRUNCATION LEVEL BE SELECTED AT 10^5 - 5.10^5 EXCEEDANCES PER 1,000 FLIGHTS, DEPENDING UPON HOW STEEP THE EXCEEDANCE CURVE IS AT ITS EXTREME POINT. THAT RECOMMENDATION IS REITERATED HERE.

5.4.5 Crack Shape

The most common crack shape in crack-growth analysis will be the quarter-circular corner flaw at the edge of a hole. Stress-intensity solutions for this case were presented in Chapter 4. For use in crack-growth analysis, these solutions present some additional problems. The stress intensity varies along the periphery of the crack. Since crack growth is a strong power function of the stress intensity, crack extension also will vary along the crack front. If this is accounted for in a calculation, the flaw shape at a hole changes from quarter circular to elliptical.

For the calculation it would be sufficient to include two points of the crack front, e.g., the crack tip at the surface and the crack tip at the edge of the hole. The stress intensity is calculated at these points, and the amount of crack growth determined. There will be a different amount of growth along the surface than along the edge of the hole. The new crack will have a size $a_i + \Delta a_s$ along the surface, and a size $a_i + \Delta a_h$ along the hole. For the next crack-growth increment the flaw may be considered a quarter-elliptical flaw with semi-axes $a_i + \Delta a_s$, and $a_i + \Delta a_h$.

There are several reasons why this procedure may still not give the accuracy expected:

-- The variation of stress intensity along a corner flaw front at the edge of a hole is not accurately known.

-- The differences in stress intensity cause differences in growth and flaw shape development. If this is so, the difference in crack-growth properties in the two directions (anisotropy) should be accounted for too.

-- The differences in growth rates and stress intensity would give also different retardation effects. There are no experimental data to support this. Fortunately, experimental evidence indicates that quarter-circular corner cracks at holes do not show significant changes in shape (55).

FOR THE FLAWS SPECIFIED IN MIL-A-83444 IT IS ACCEPTABLE AT THIS POINT IN TIME TO IGNORE FLAW SHAPE CHANGES. CRACK GROWTH MAY BE ASSUMED TO BE THE SAME ALL ALONG THE CRACK FRONT, SUCH THAT THE FLAW REMAINS QUARTER-CIRCULAR. Consequently, crack growth needs to be calculated at one point only. It is recommended that the same point be taken as is used for the evaluation of residual strength (see Chapter 4).

When the flaw size has become equal to the plate thickness, the flaw will become a through crack with a curved front for which stress-intensity solutions are readily available. Cracks usually have a tendency to quickly become normal through cracks once they reach the front free surface (Figure 5-43). THEREFORE, IT IS RECOMMENDED TO CONSERVATIVELY ASSUME THE CRACK TO BECOME A NORMAL THROUGH CRACK OF A SIZE EQUAL TO THE THICKNESS IMMEDIATELY AFTER IT REACHES THE FREE SURFACE (a = B, FIGURE 5-43).
More realistic flaw development assumptions can be made if there is experimental

support for the adequacy of the crack-growth predictions made.

5.4.6 Interaction of Cracks

Regarding initial flaw assumptions, para. 3.1.1 of MIL-A-83444 states:

> "Only one initial flaw in the most critical hole and one initial flaw at a location other than a hole need be assumed to exist in any structural element. Interaction between these assumed initial flaws need not be considered."

Obviously, interaction between these cracks can be disregarded because these cracks are not assumed to occur simultaneously, although each of them may occur separately. However, more than one initial flaw may occur if due to fabrication and assembly operations two or more adjacent elements can contain the same initial damage at the same location. Note that each of the adjacent elements has only one flaw. Para. 3.1.1 of MIL-A-83444 states:

> "For multiple and adjacent elements, the initial flaws need not be situated at the same location, except for structural elements where fabrication and assembly operations are conducted such that flaws in two or more elements can exist at the same location."

The previous statement that interaction between assumed initial flaws need not be considered is not repeated here because these cracks will interact as they occur simultaneously. In principle, the damage tolerance calculation should consider this interaction. However, a rigorous treatment of this problem is prohibitive in most cases. Consider, e.g., a skin with a reinforcement as in Figure 5-44. Because of assembly drilling both holes should be assumed flawed (Figure 5-44a). If both elements carry the same stress, there will be hardly any load transfer initially. Hence, the stress intensities for both flaws will be equal, implying that initially both will grow at the same rate.

If the two cracks continue to grow simultaneously in a dependent manner, their stress intensities will eventually be different (e.g, K of the reinforcement would increase faster if only for the finite size effect). This means that in a given cycle the rate of growth would be different for the two cracks resulting in different crack sizes. Since it cannot be foreseen a priori how the crack sizes in the two members develop, it would be necessary to develop K-solutions for a range of crack sizes and a range of crack size ratios in the two members.

Assume the crack size in the skin is a_s, the crack size in the reinforcement a_r. For a given value of a_r, the K for the skin crack would be calculated as a function of a_s. This calculation would be repeated for a range of a_r sizes. The same would be done for the reinforcement crack and a range of a_s values. For any given combination of a_r and a_s, the two stress intensities then can be found by interpolation.

Although the consequences of crack interaction should be evaluated, routine calculations may be run without interaction of cracks (56,57). Obviously, the calculation procedure is much simpler if interaction can be ignored. However, the procedure may give unconservative results.

If either element remained uncracked, the stress intensity in the other element would be much lower, becuase there would be load transfer from the cracked element to the uncracked element. Obviously, the stress intensity in the skin would be the lowest. The cracks could be grown as if the other element were uncracked and crack growth would be slower.

Finally, the reinforcement would be totally cracked. From there on interaction must be taken into account, i.e., the crack in the skin would be treated now for the case of a failed reinforced panel (e.g., stringer reinforced structure with middle stringer failed).

This means that two analyses have to be made for a K-determination, one with the reinforcement uncracked, one with the reinforcement failed. If the two independent crack-growth analyses show that the reinforcement has failed, the analysis of the skin is changed appropriately.

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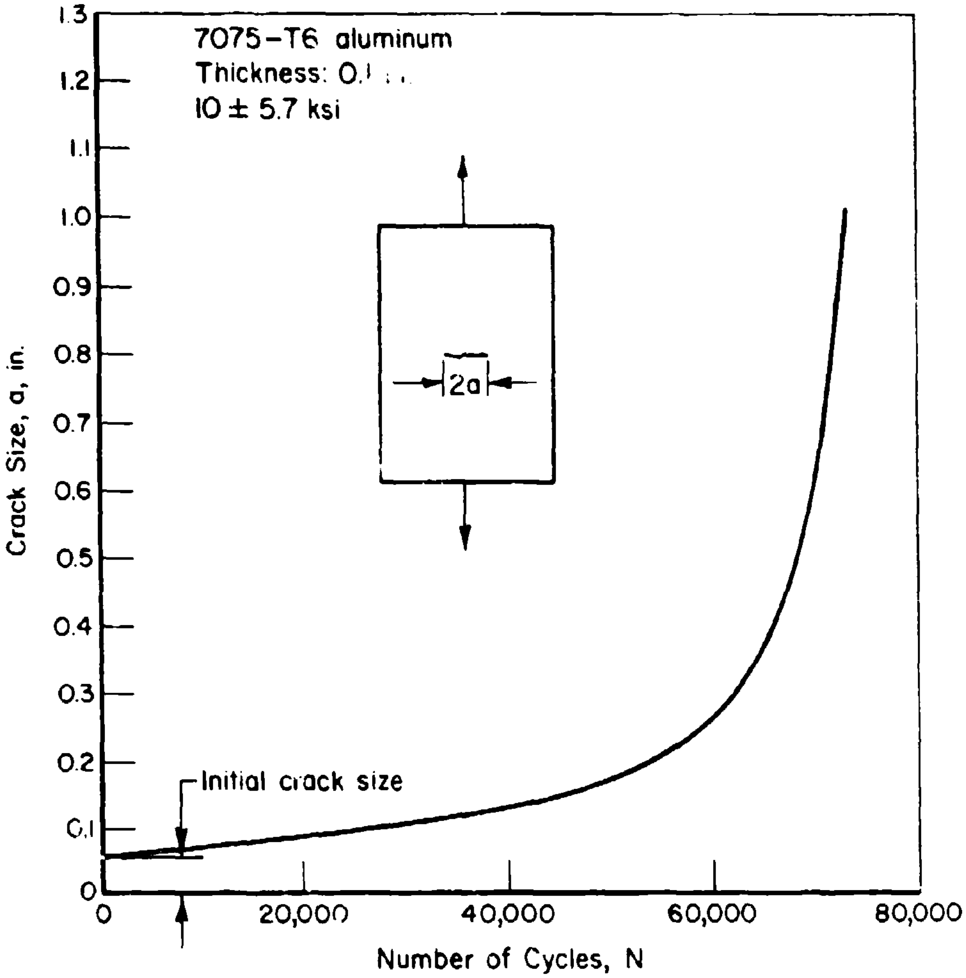


FIGURE 5.1 TYPICAL CRACK GROWTH CURVE

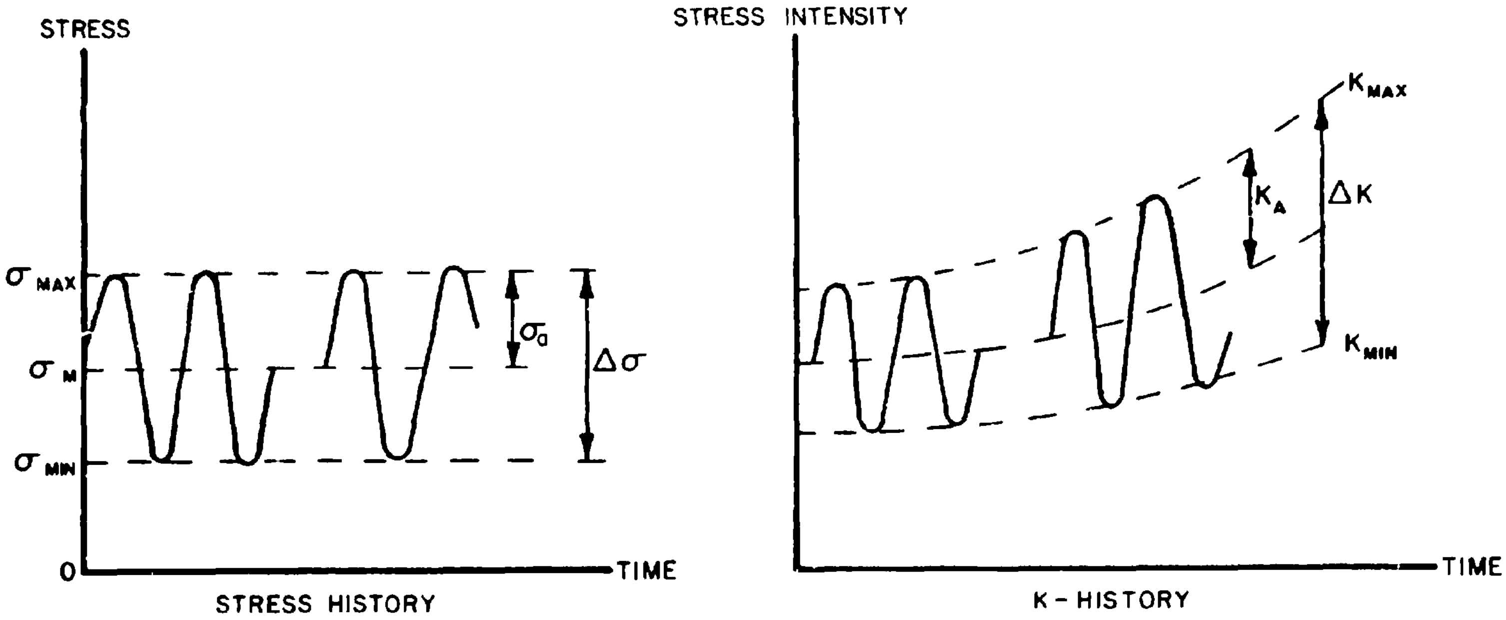


FIGURE 5.2 DEFINITION OF TERMS

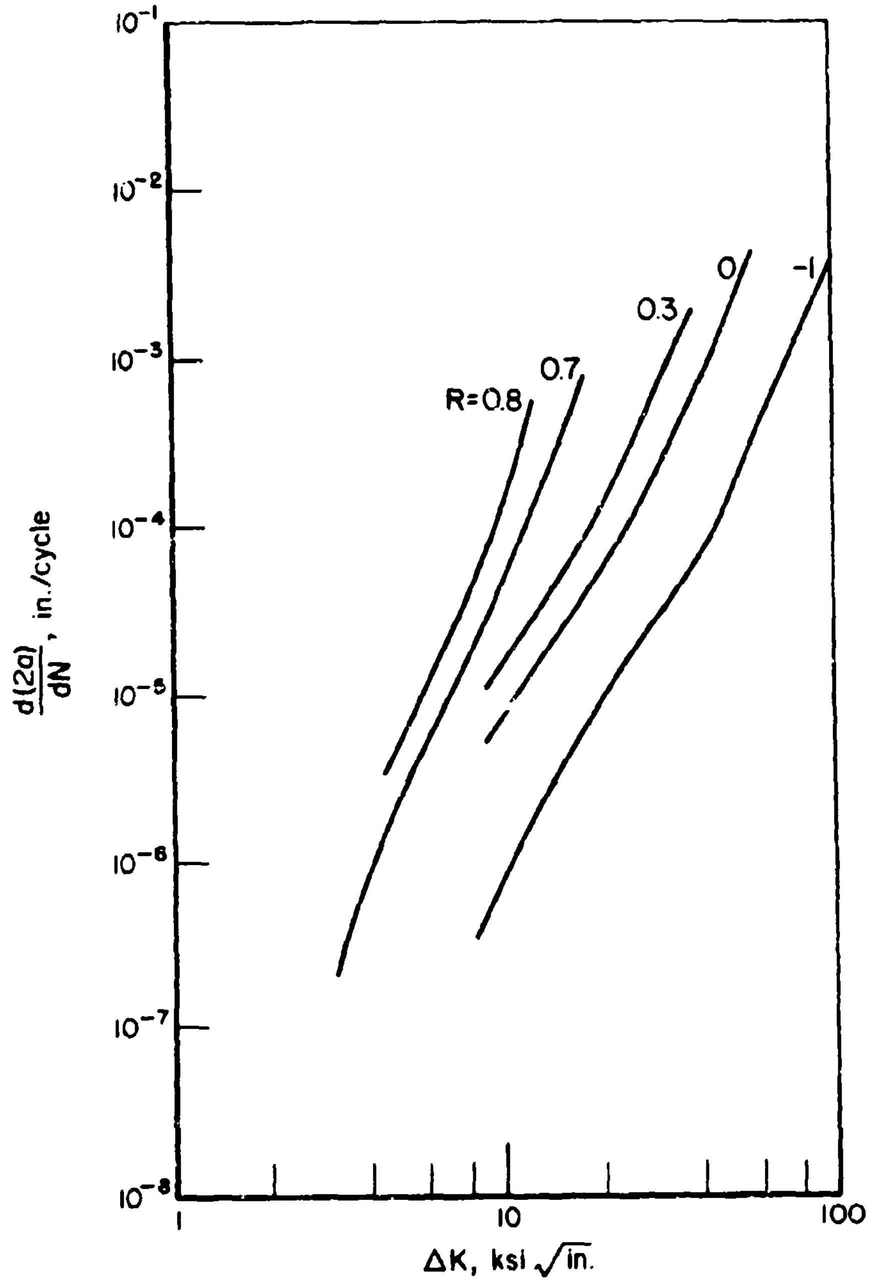


Figure 5.3 Crack Growth Rate Trends for an Aluminum Alloy Illustrating the Effect of Stress Ratio

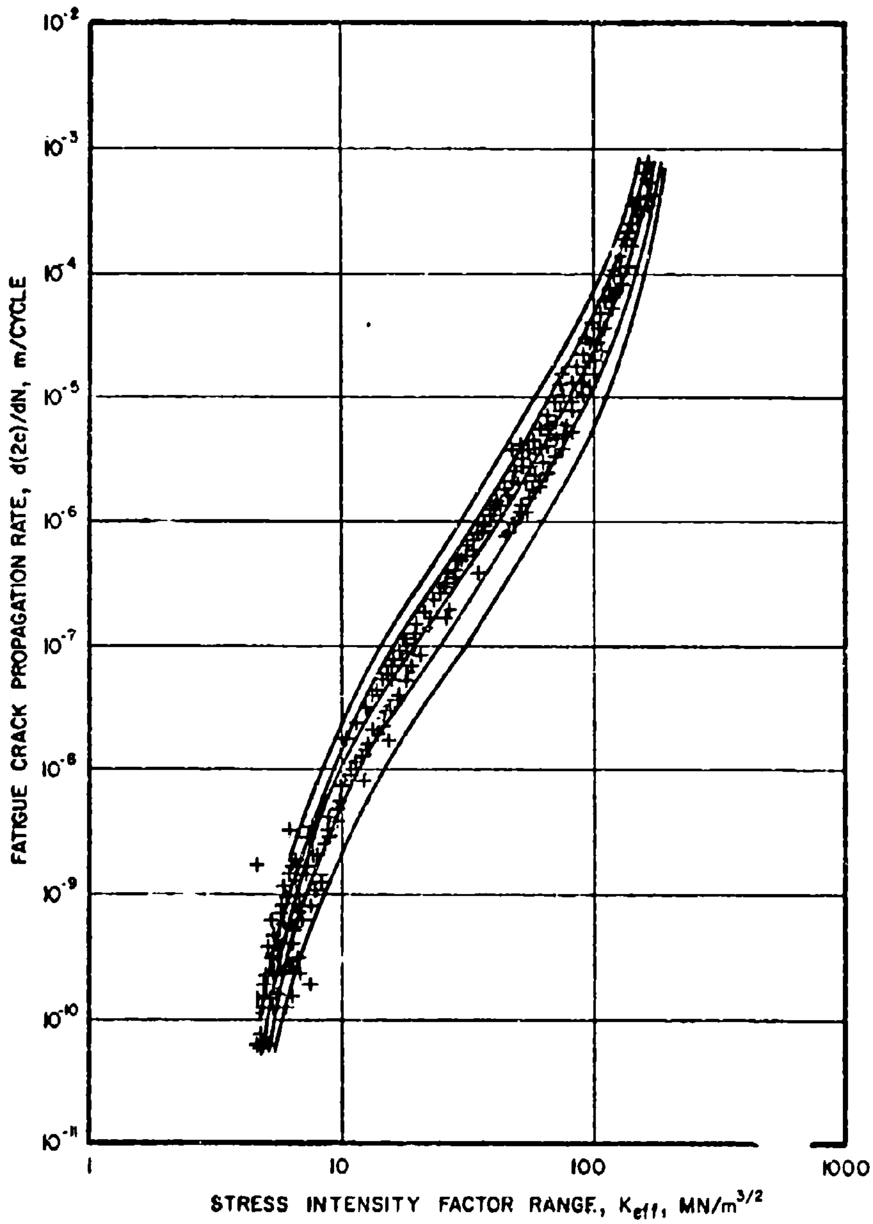
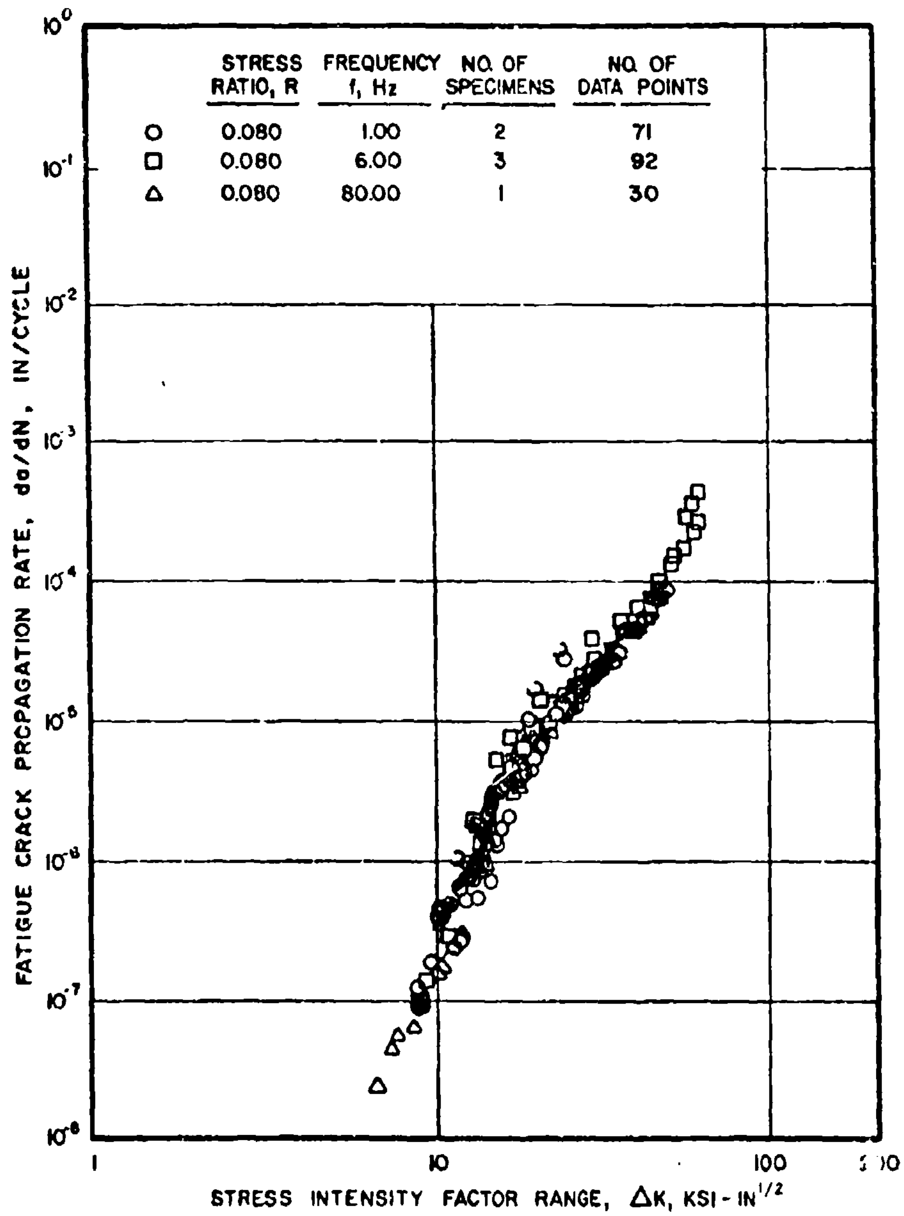


FIGURE 5.4 FATIGUE-CRACK-PROPAGATION-RATE CURVE FOR Ti-6AL-4V ALLOY



TI-6Al-4V, 1.50 IN. RA PLATE, CT SPECIMENS, L-T DIRECTION
TYS: 121, 118; TUS: 135, 129
ENVIRONMENT: 70 F, LOW HUM. AIR
SPECIMEN THK.: 0.99; WIDTH: 7.40 IN.
REFERENCE NO.: 85837, 88579

FIGURE 5.5 EXAMPLE PAGE OF DAMAGE TOLERANCE DATA HANDBOOK

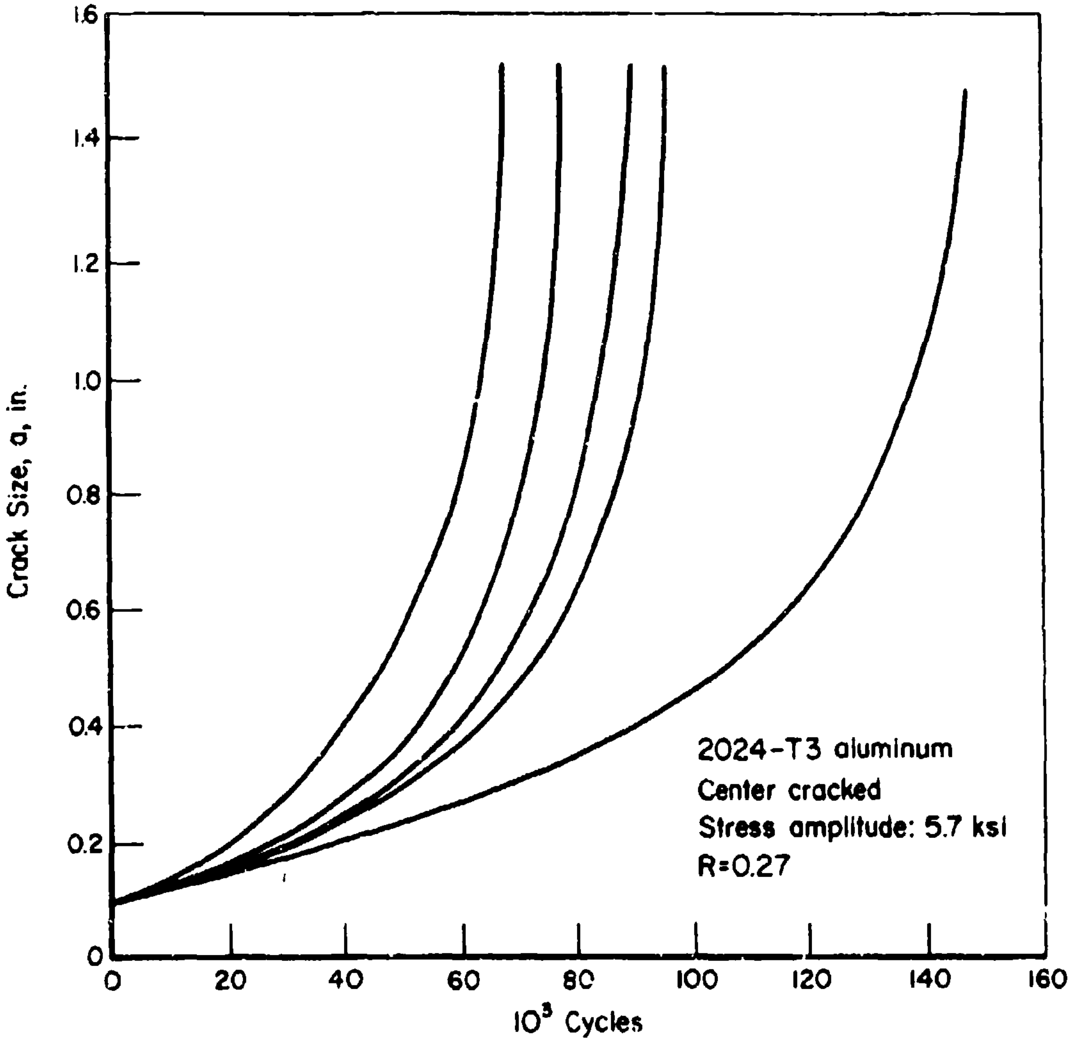


FIGURE 5.6 POSSIBLE VARIATION OF CRACK GROWTH IN MATERIALS FROM DIFFERENT SOURCES (Ref. 10)

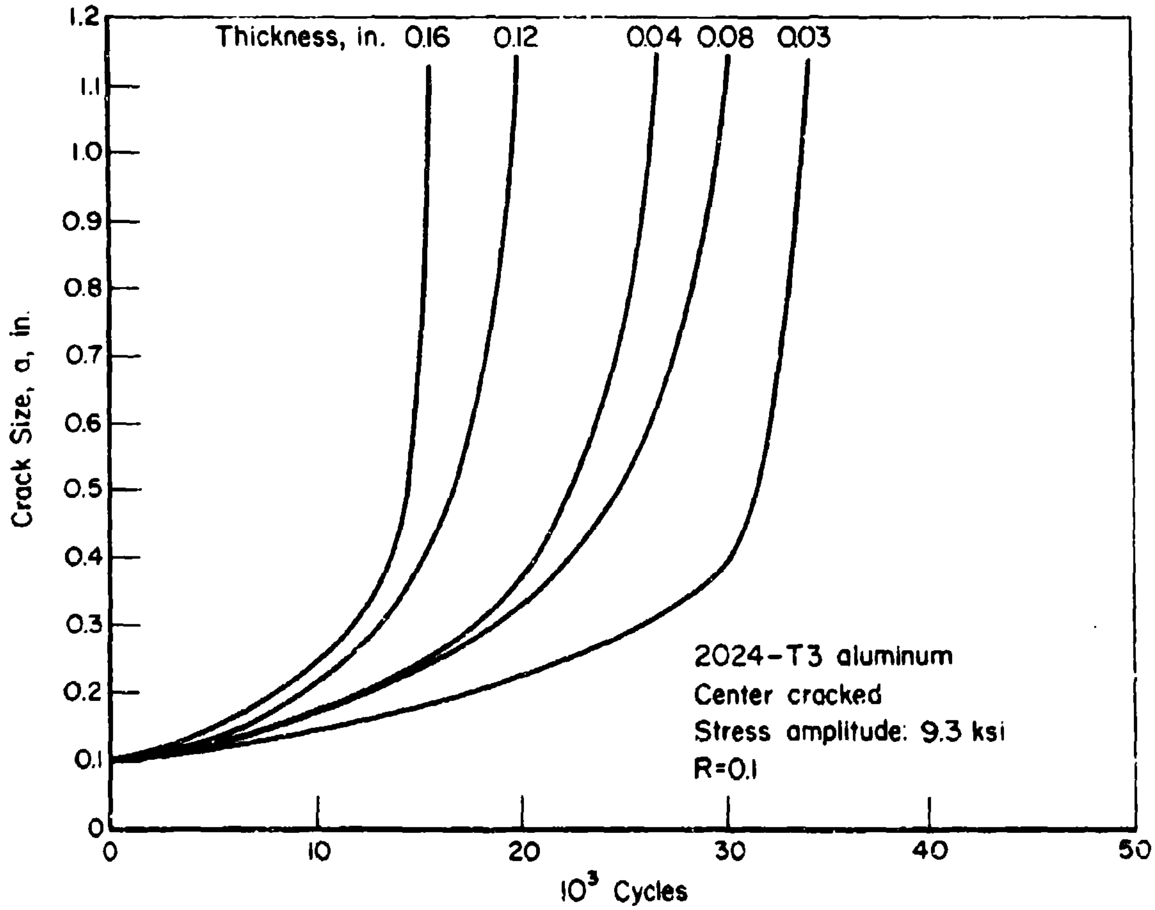


FIGURE 5.7 EXAMPLE OF EFFECT OF THICKNESS ON CRACK GROWTH (Ref. 11)

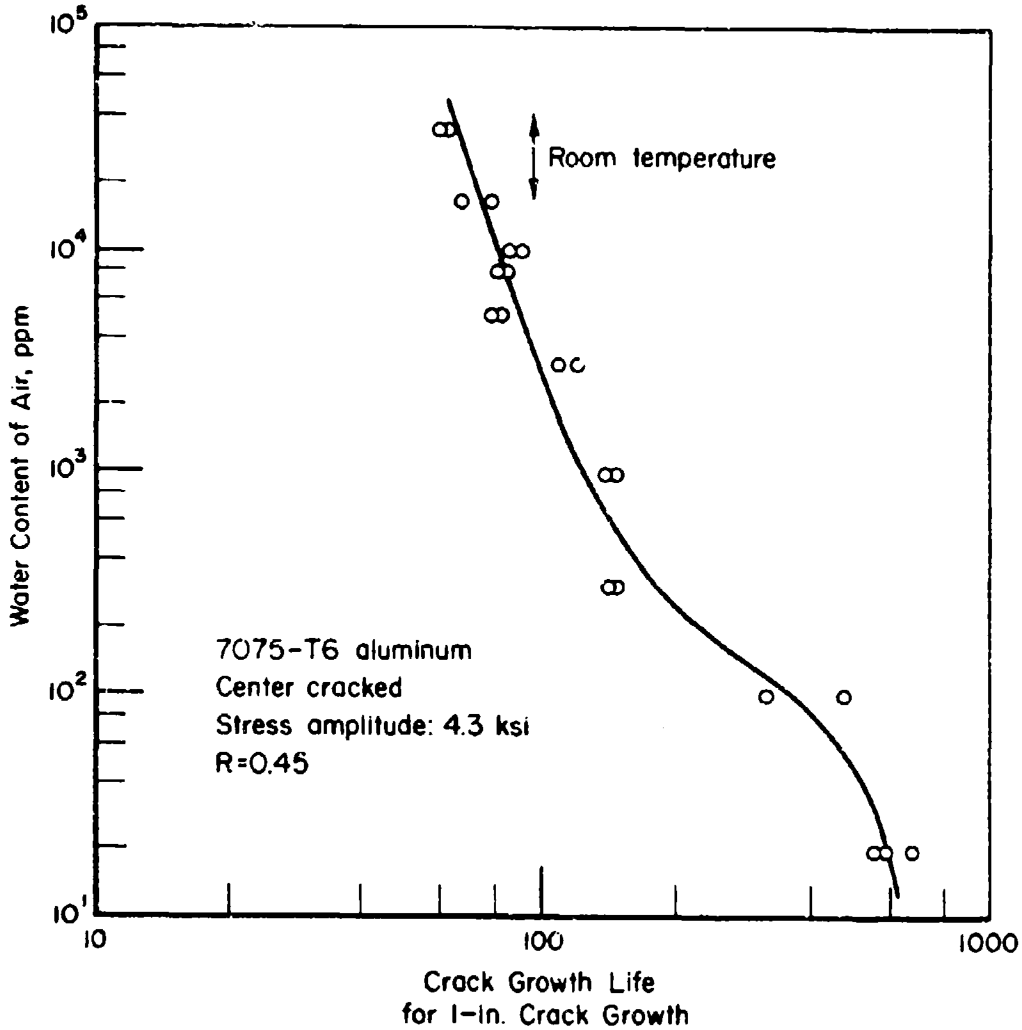


FIGURE 5.8 EFFECT OF HUMIDITY ON FATIGUE CRACK PROPAGATION (Ref. 16)

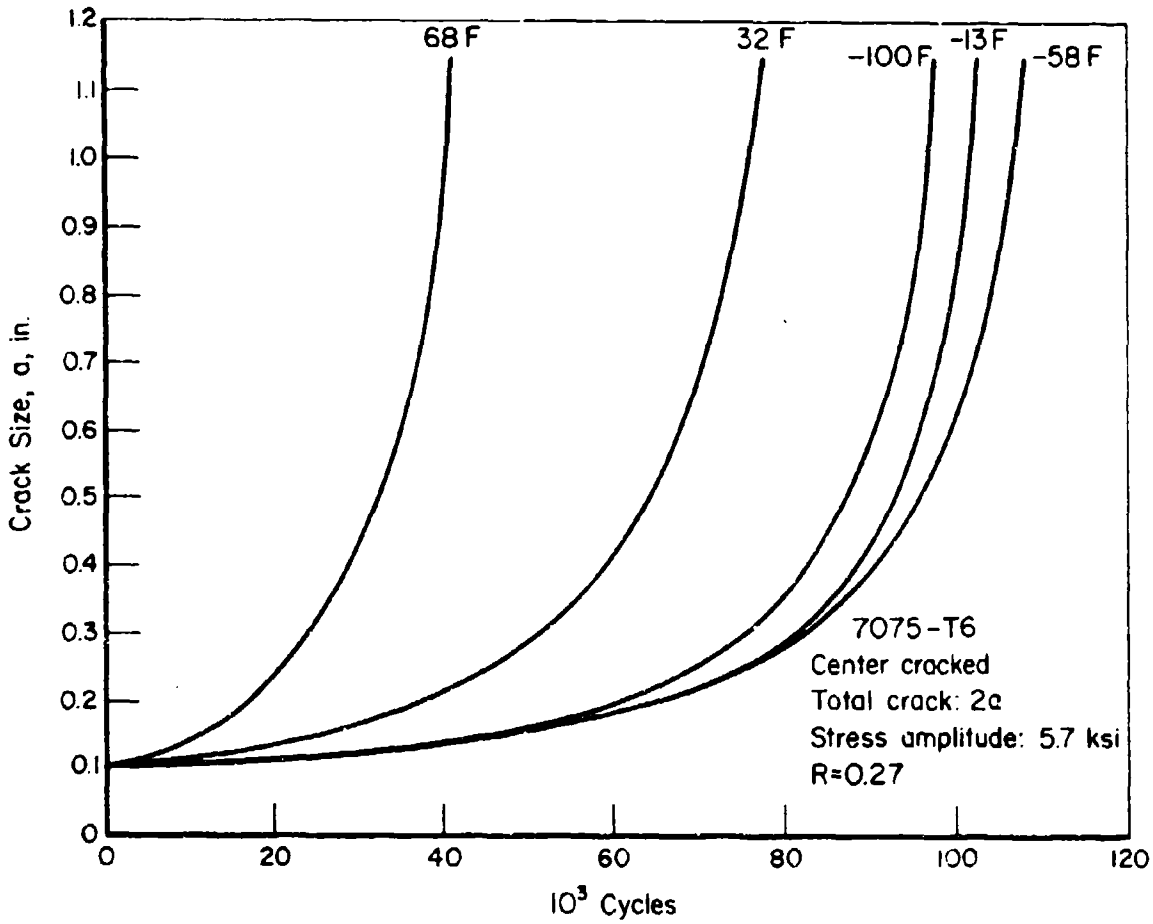


FIGURE 5.9 EXAMPLE OF TEMPERATURE EFFECT ON CRACK GROWTH (Ref. 27)

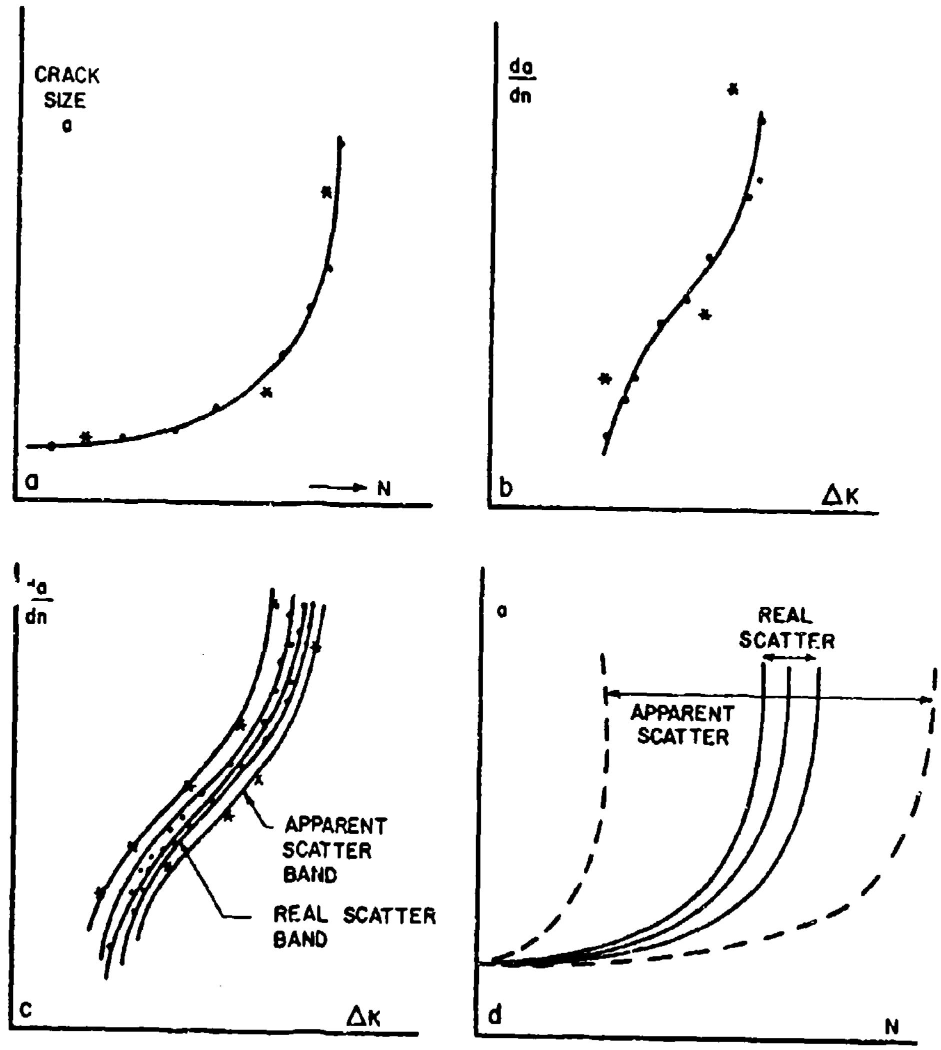
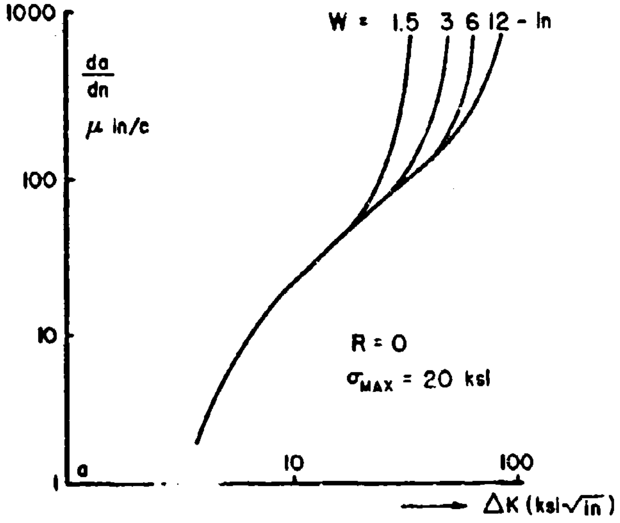


FIGURE 5.10 ANALYSIS OF CRACK GROWTH DATA



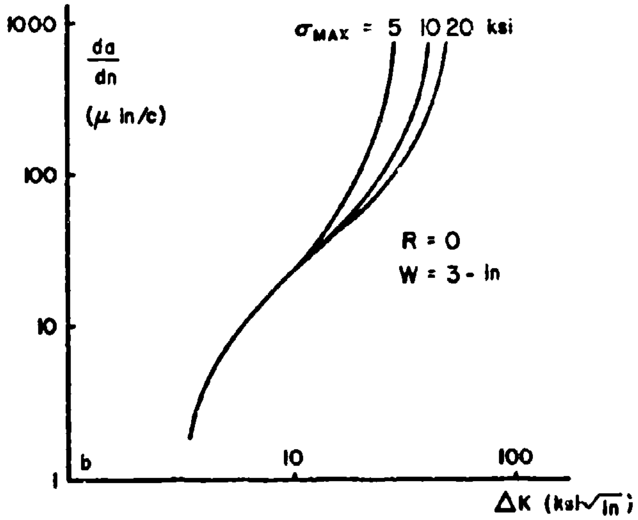


FIGURE 5.11 EFFECT OF PANEL SIZE ON FATIGUE CRACK GROWTH

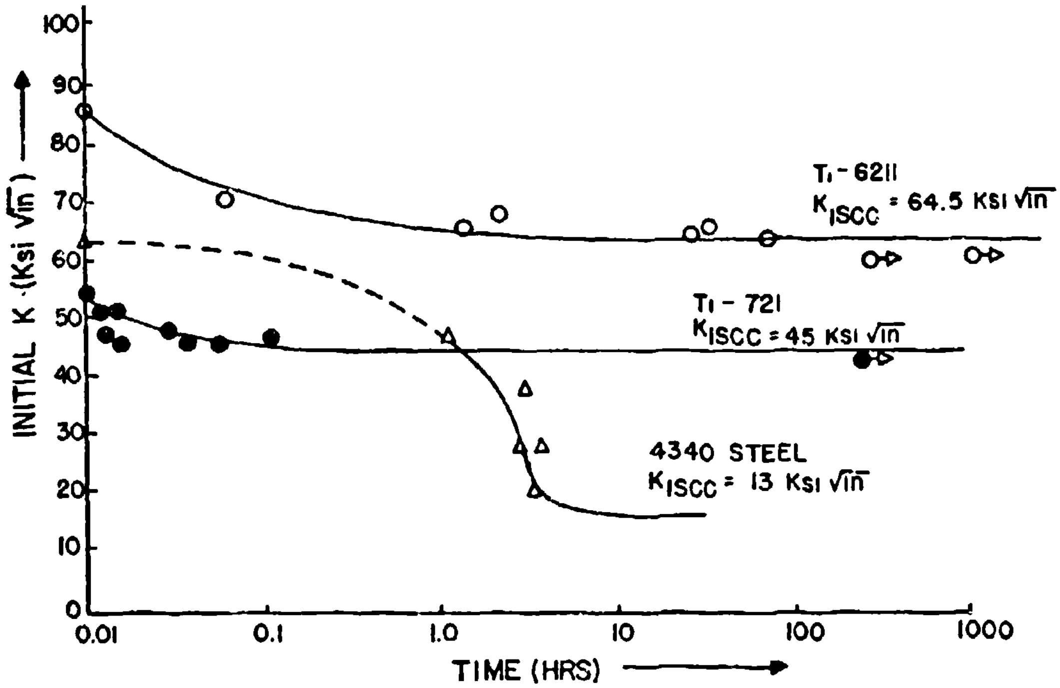


Figure 5.12 Stress Corrosion Cracking Data

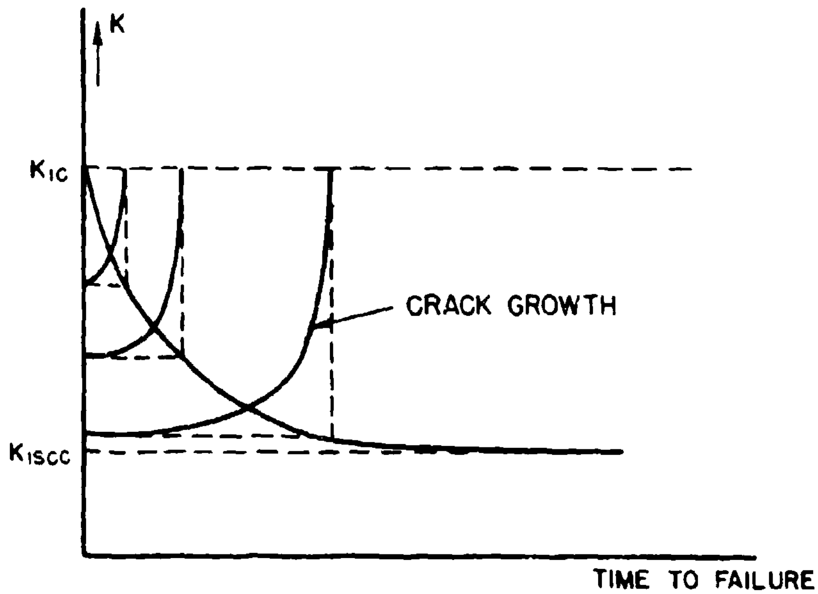


Figure 5.13 Stress Corrosion Cracking

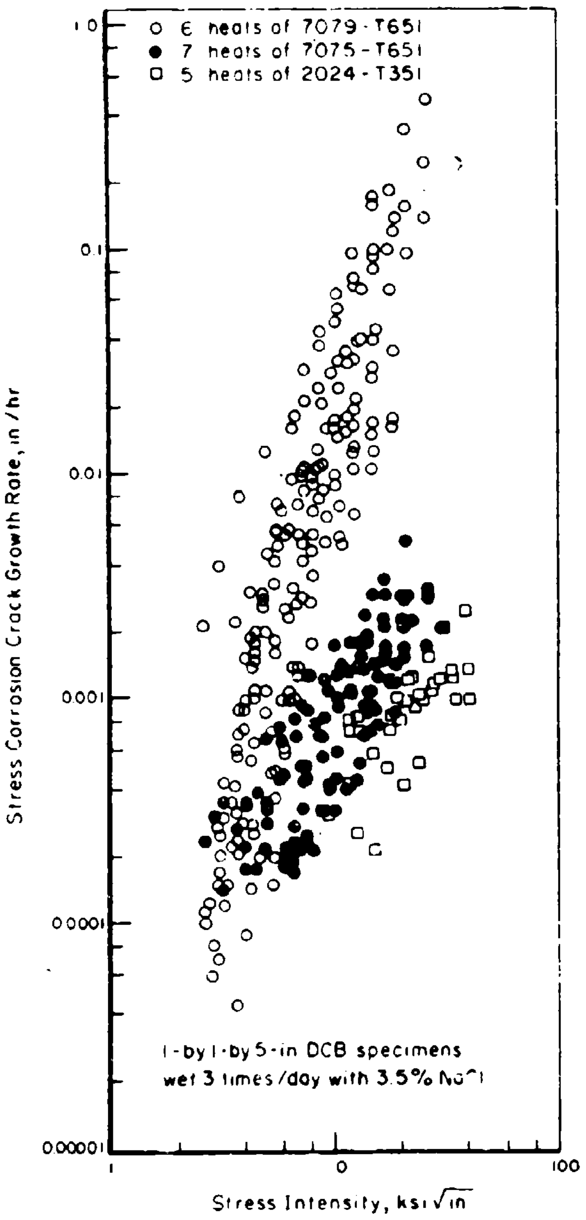


Figure 5.14 Sustained Load Crack-Growth Rate Data for 7075-T651, 7079-T651, and 2024-T351 Aluminum Plate (Ref. 9.)

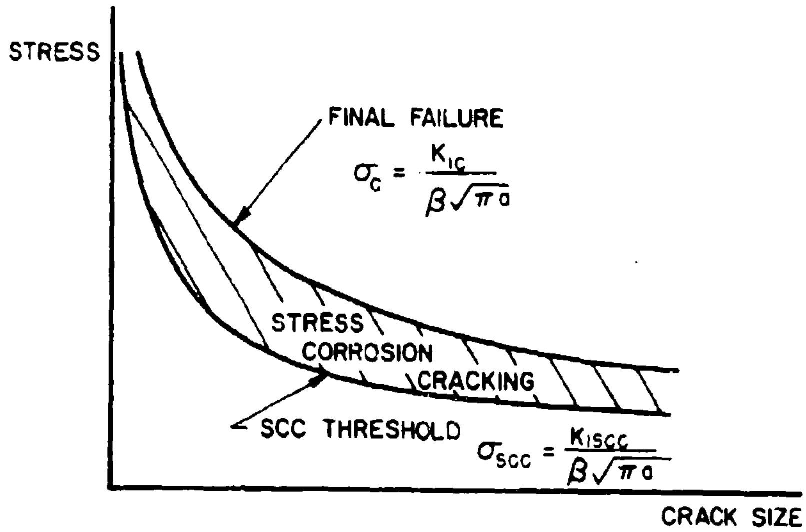


Figure 5.15 Stress Required for Stress Corrosion Cracking

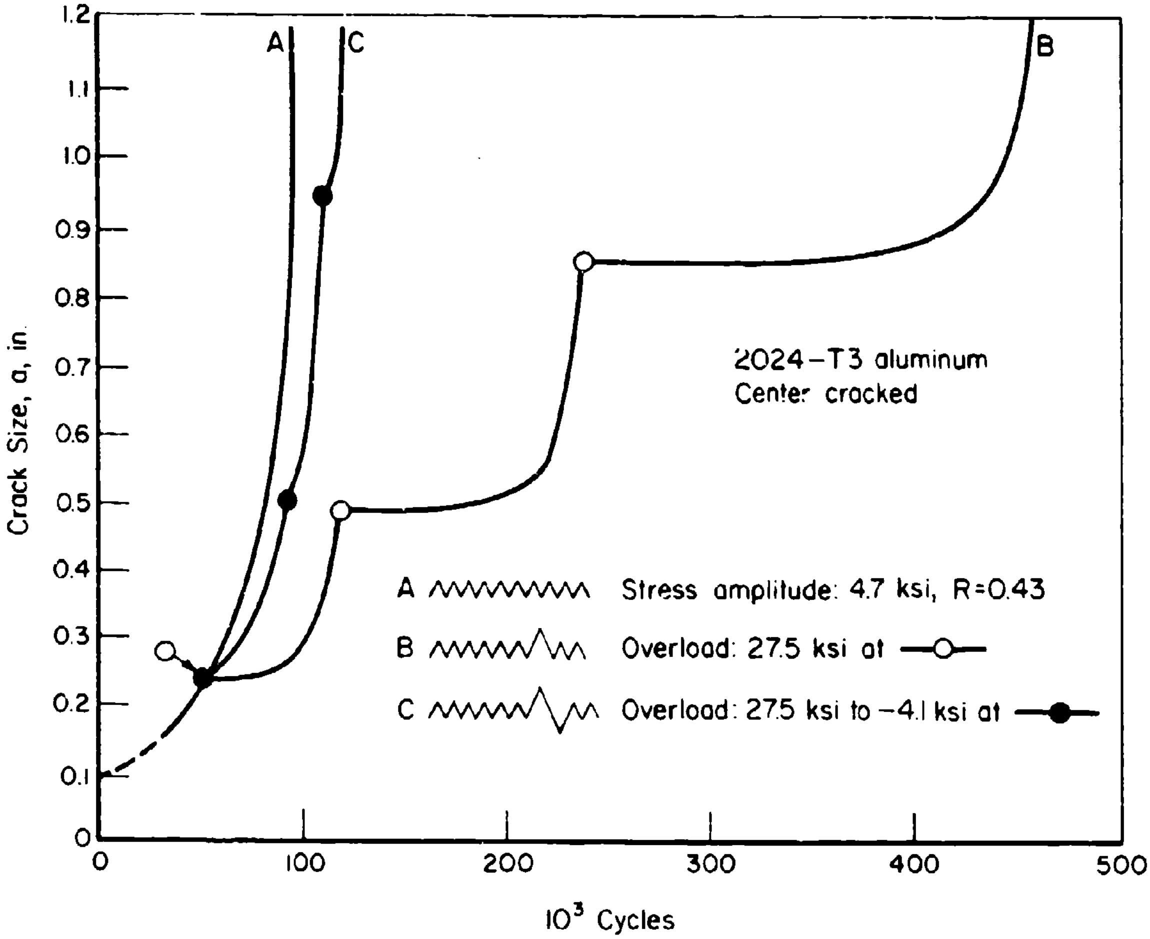


FIGURE 5.16 RETARDATION DUE TO POSITIVE OVERLOADS, AND DUE TO POSITIVE-NEGATIVE OVERLOAD CYCLES (Ref. 34)

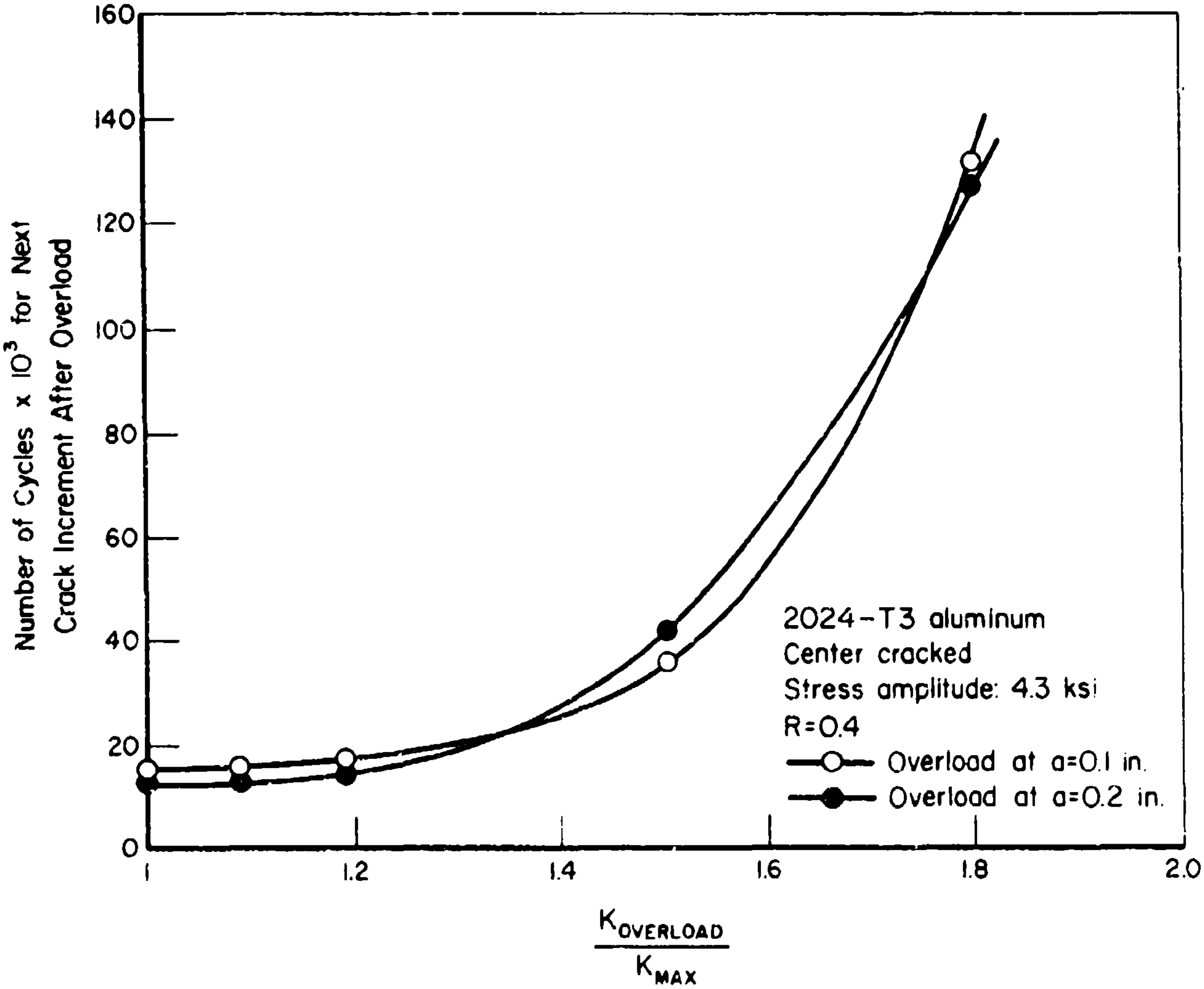


FIGURE 5.17 EFFECT OF MAGNITUDE OF OVERLOAD ON RETARDATION

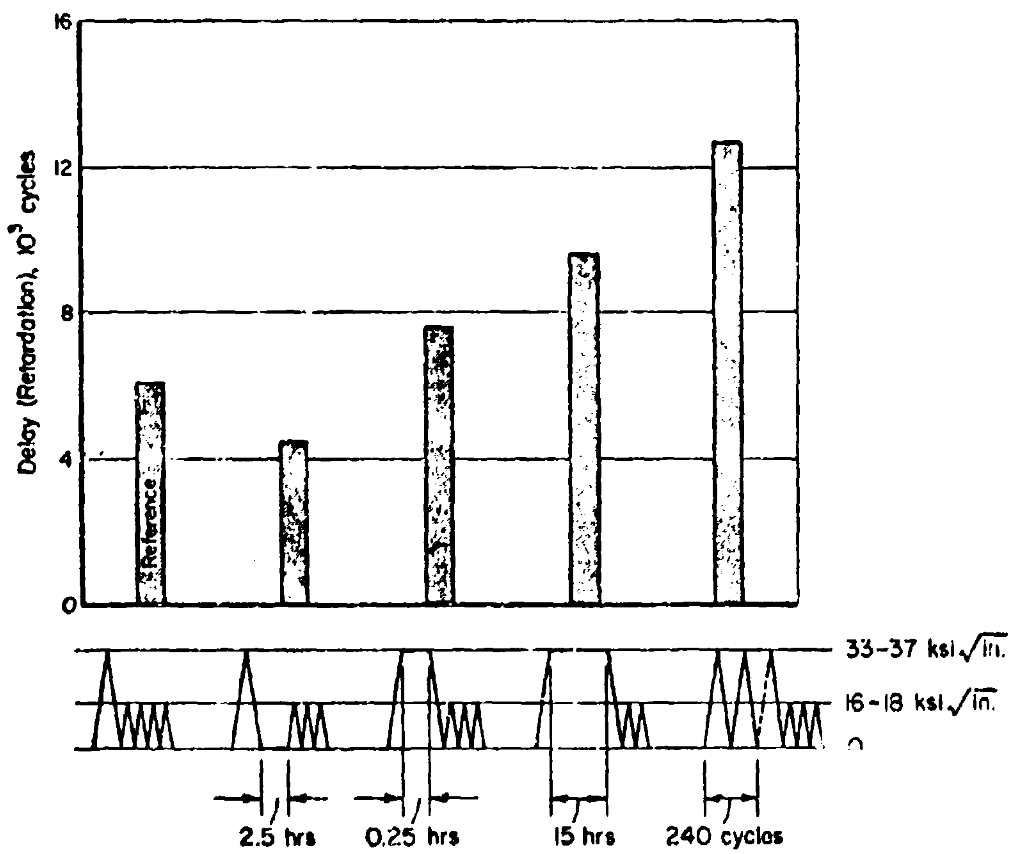


FIGURE 5.18 RETARDATION IN Ti-6V-4Al; EFFECT OF HOLD PERIODS (Ref. 37)

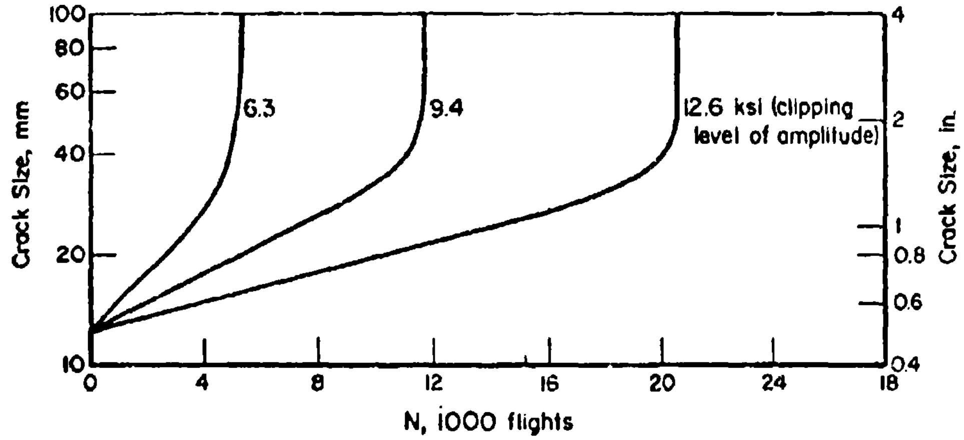


FIGURE 5.19 EFFECT OF CLIPPING OF HIGHEST LOADS IN RANDOM FLIGHT-BY-FLIGHT LOADING ON CRACK PROPAGATION IN 2024-T3 Al ALLOY (Ref 38, 39)

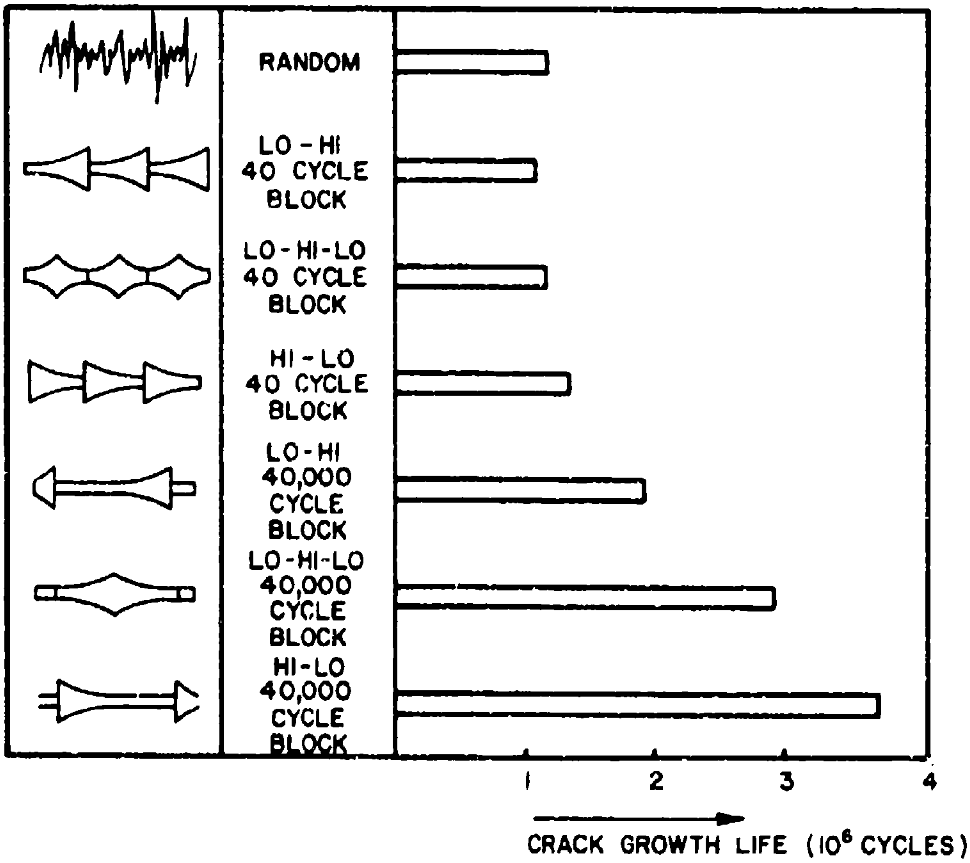


FIGURE 5.20 EFFECT OF BLOCK PROGRAMMING AND BLOCK SIZE ON CRACK GROWTH LIFE (ALL HISTORIES HAVE SAME CYCLE CONTENT) ALLOY: 2024-T3 ALUMINUM (REF. 36)

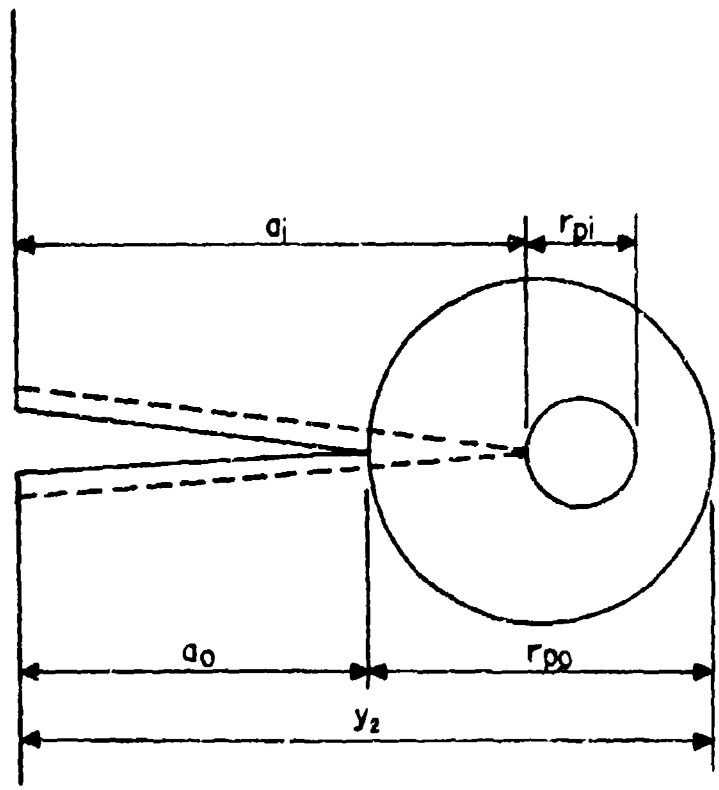


Figure 5.21 Yield Zone Due to Overload (r_{po}), Current Crack Size (a_i), and Current Yield Zone (r_{pi})

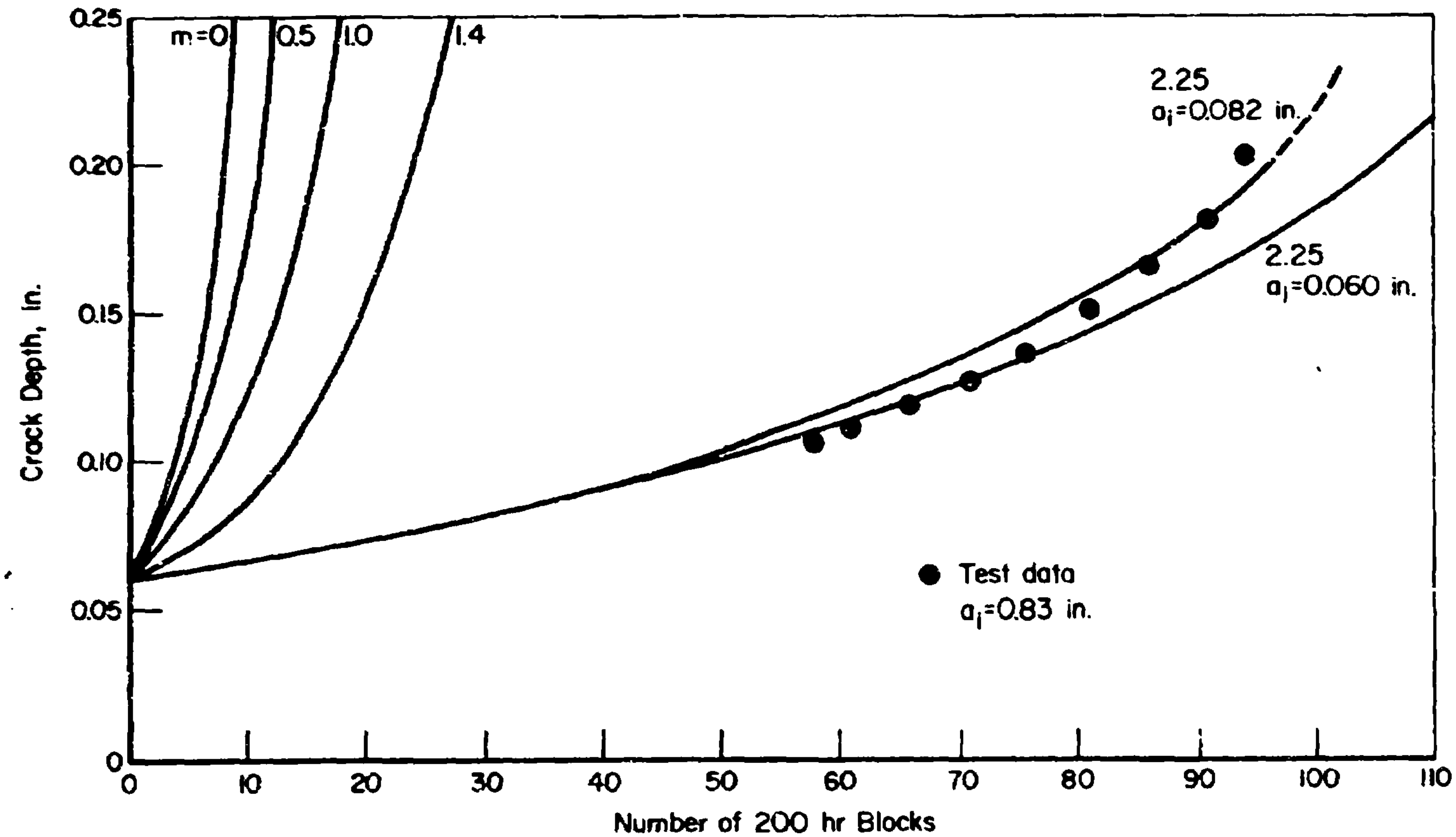


FIGURE 5.22 CRACK GROWTH PREDICTIONS BY WHEELER MODEL USING DIFFERENT RETARDATION EXPERIMENTS (Ref. 40)

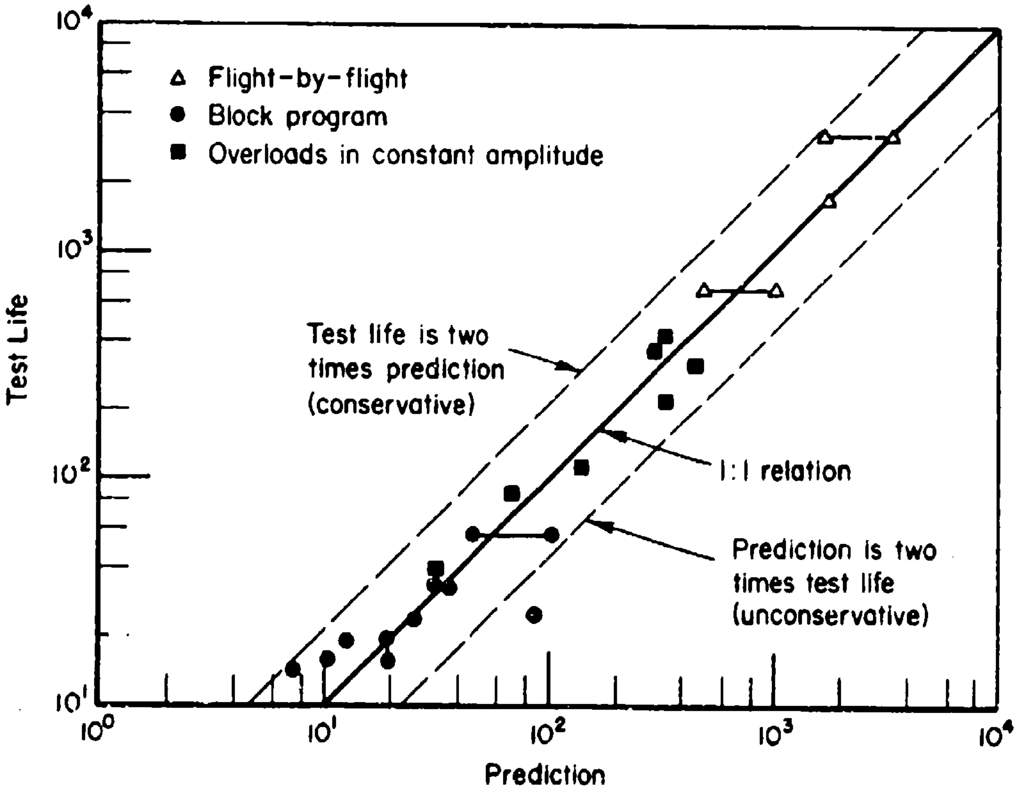


FIGURE 5.23 PREDICTIONS OF CRACK GROWTH LIVES WITH THE WILLENBORG MODEL, AND COMPARISON WITH TEST DATA (Ref. 50)

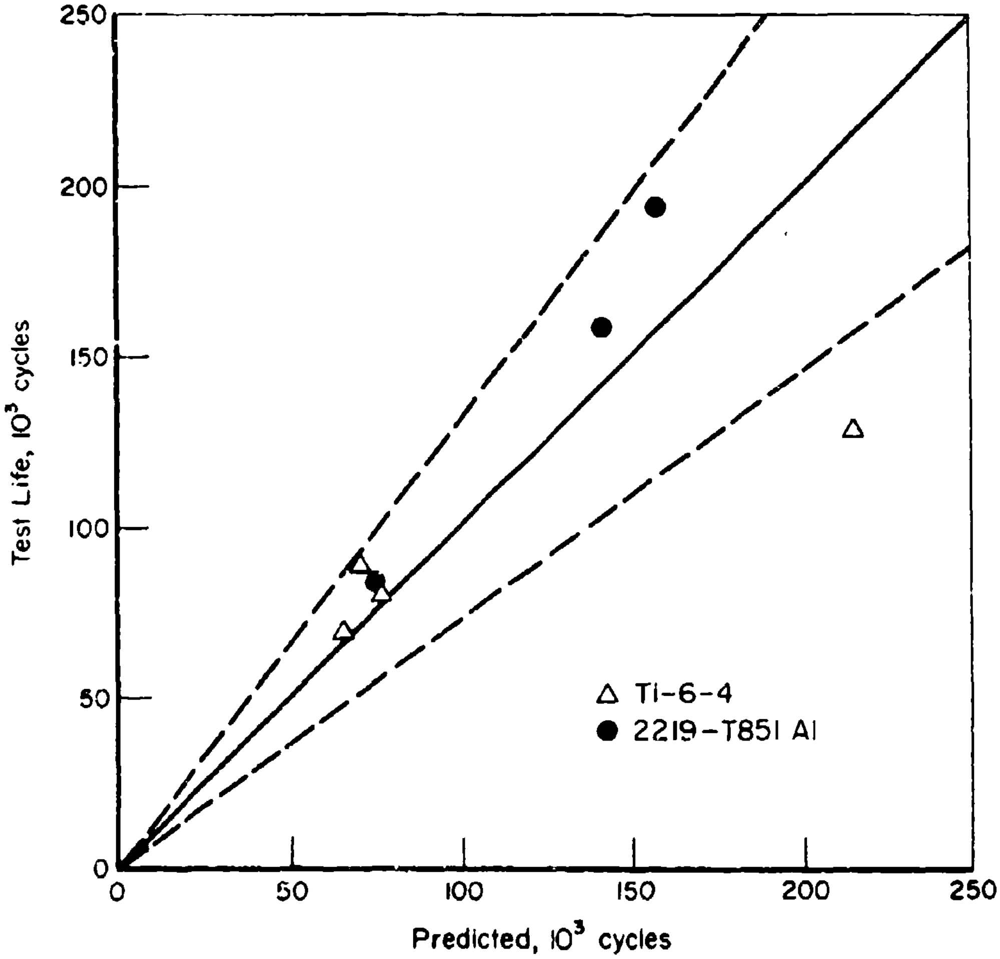


FIGURE 5.24 PREDICTIONS BY CRACK CLOSURE MODEL AS COMPARED WITH DATA OF CONSTANT AMPLITUDE TESTS WITH OVERLOAD CYCLES (Ref. 47)

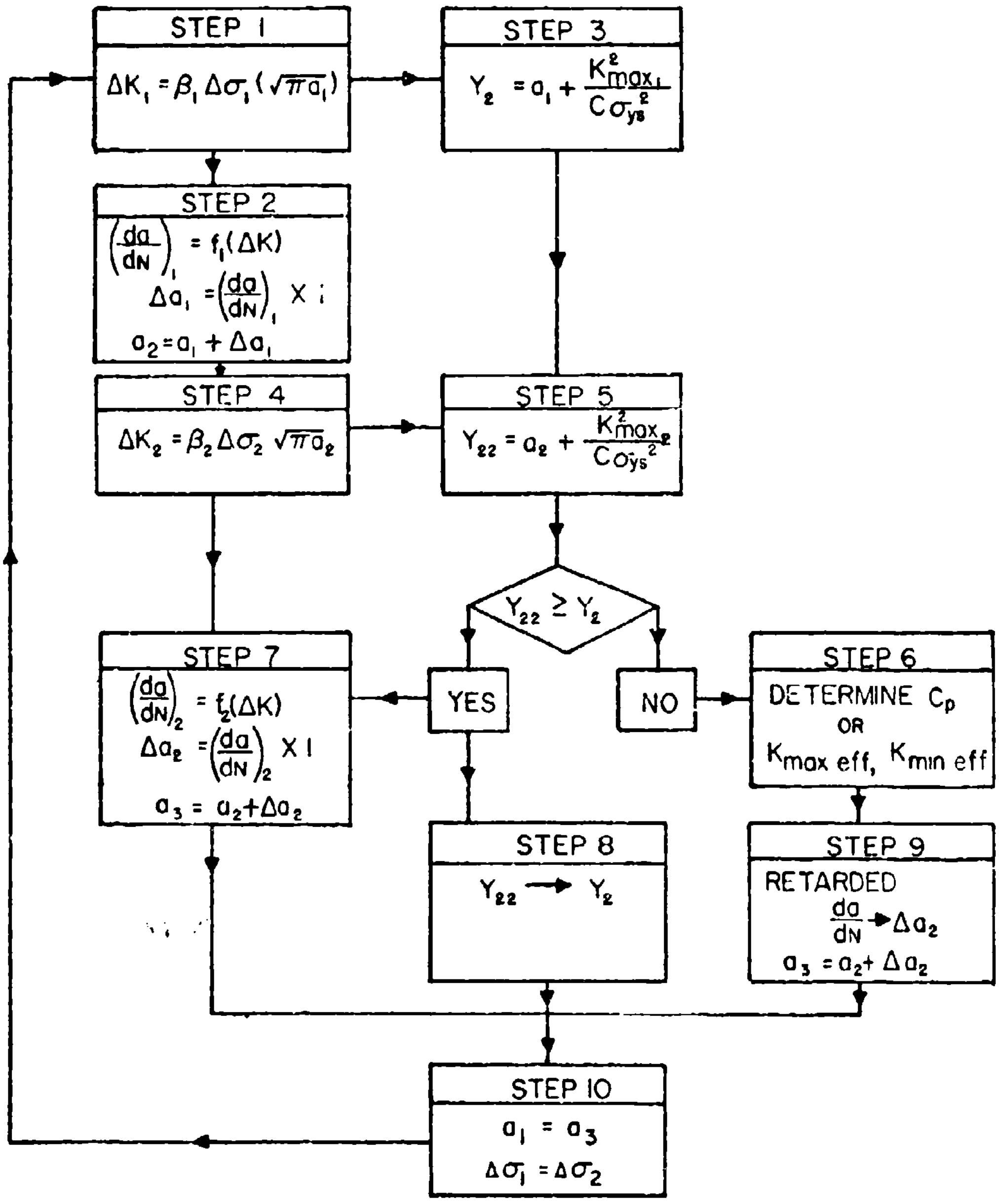
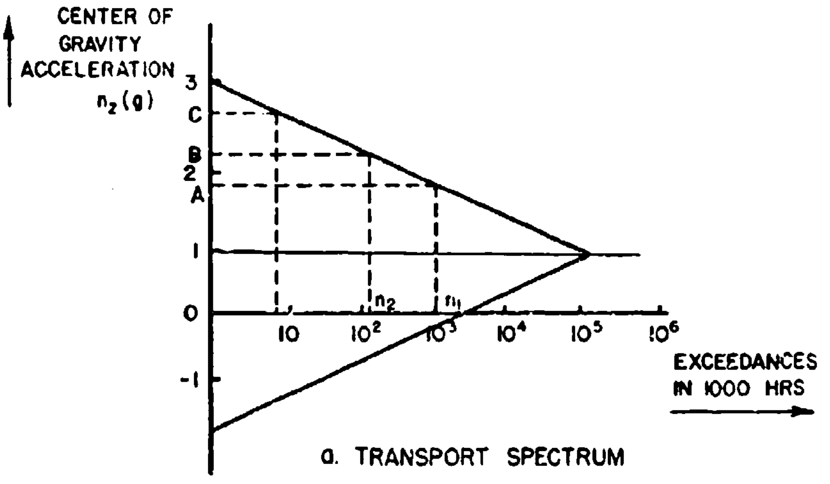
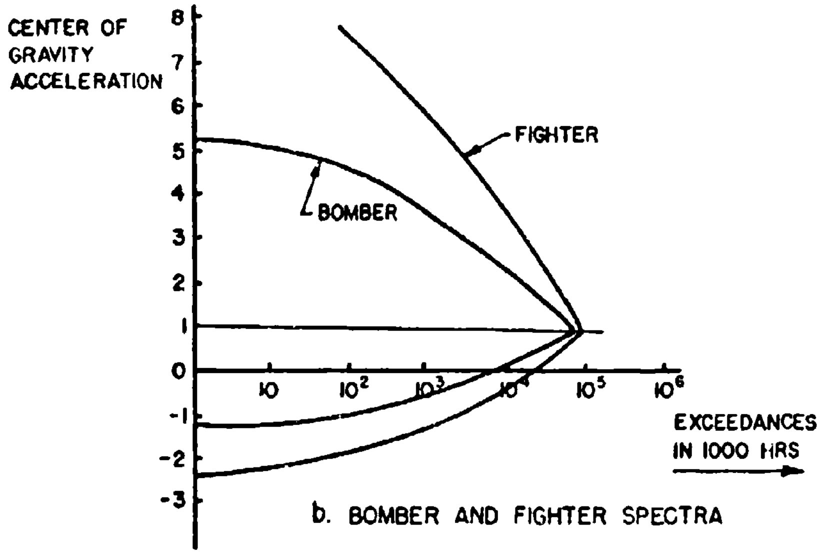


Figure 5.25 Steps Required for Crack-Growth Integration



a. TRANSPORT SPECTRUM



b. BOMBER AND FIGHTER SPECTRA

FIGURE 5.26 EXCEEDANCE SPECTRA FOR 1000 HRS.

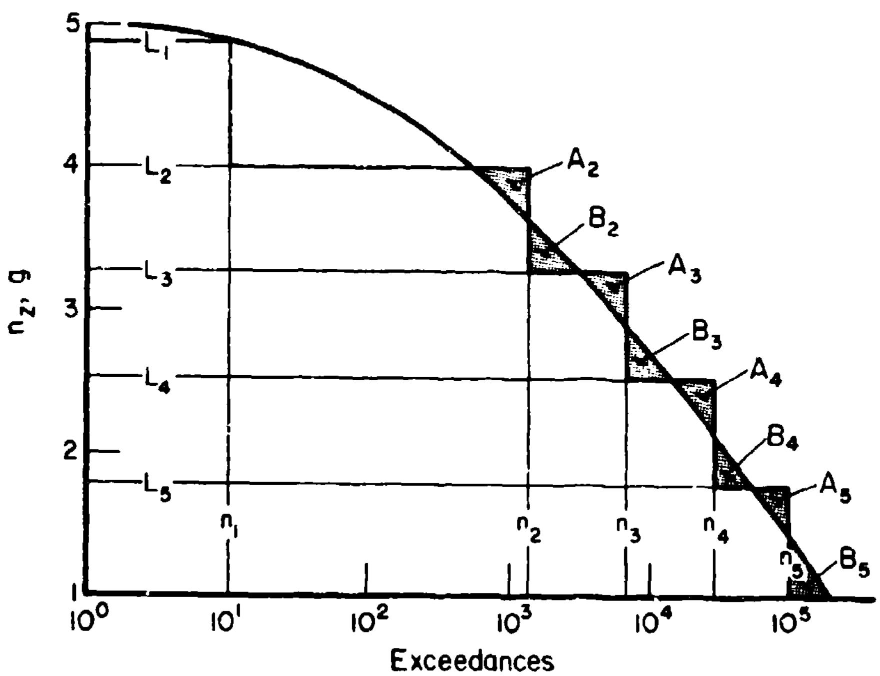


FIGURE 5.27 STEPPED APPROXIMATION OF SPECTRUM

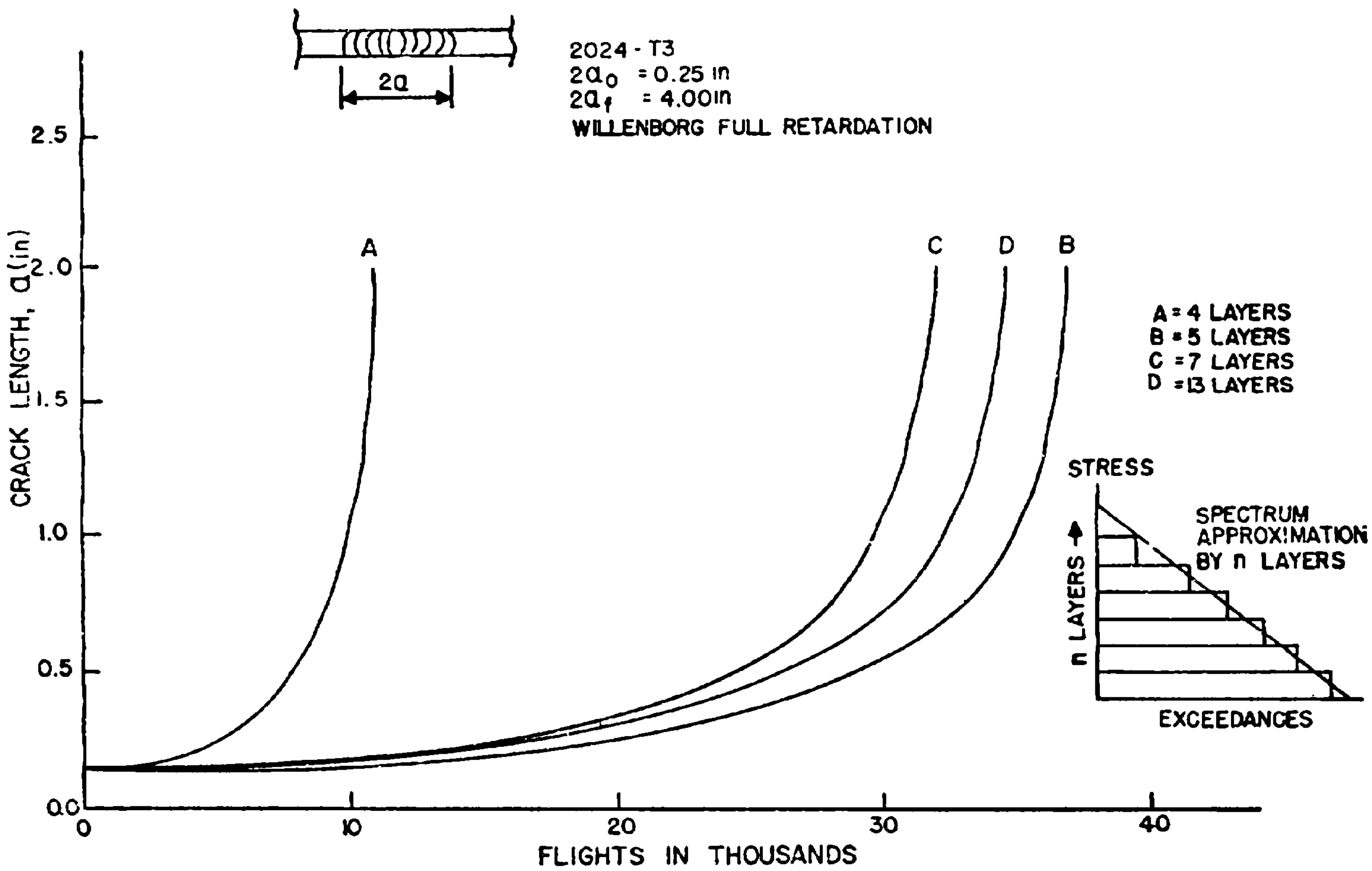


Figure 5.28 Fatigue-Crack Growth Behavior Under Various Spectra Flight-by-Flight

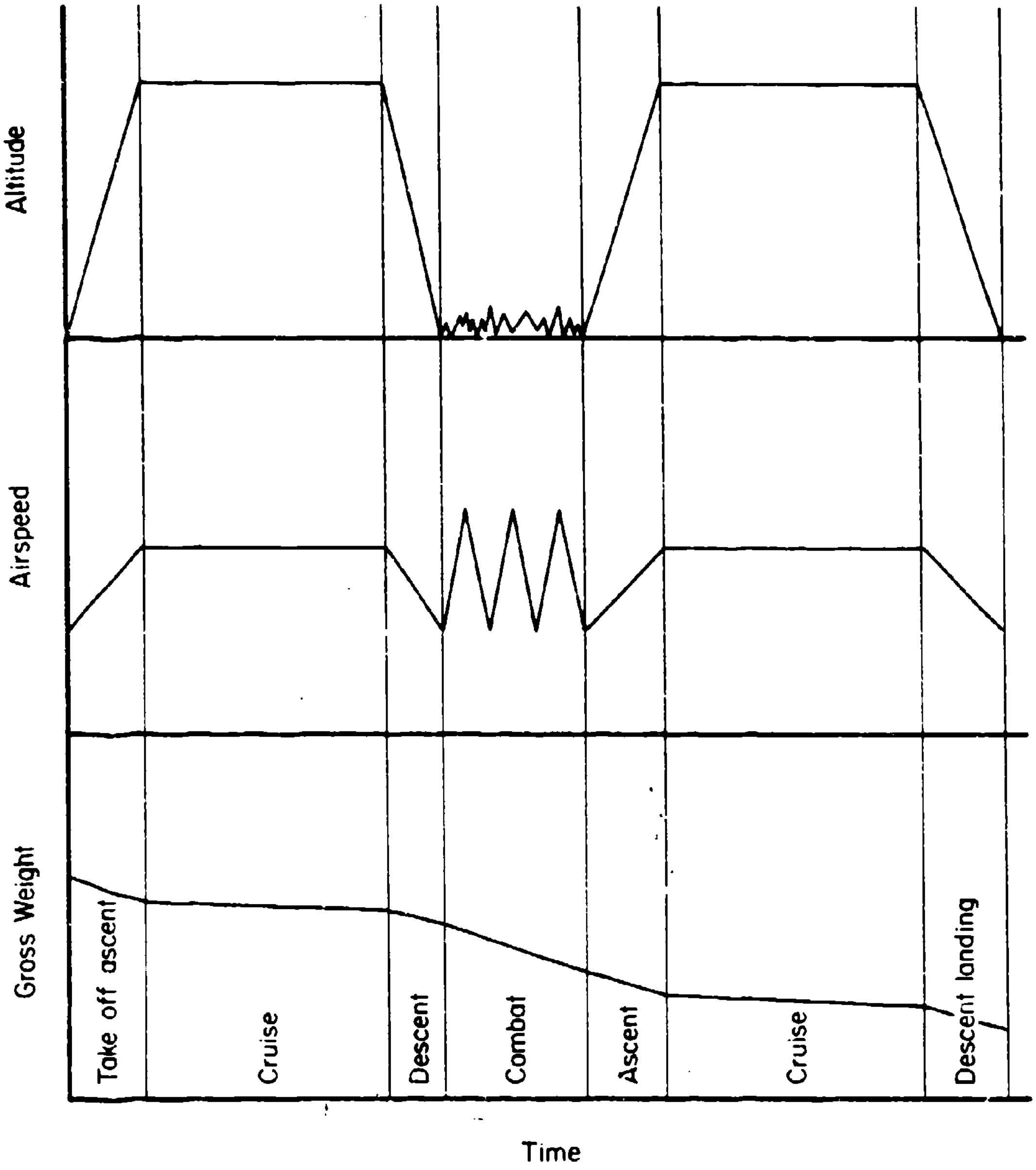


FIGURE 5.29 MISSION PROFILE AND MISSION SEGMENTS

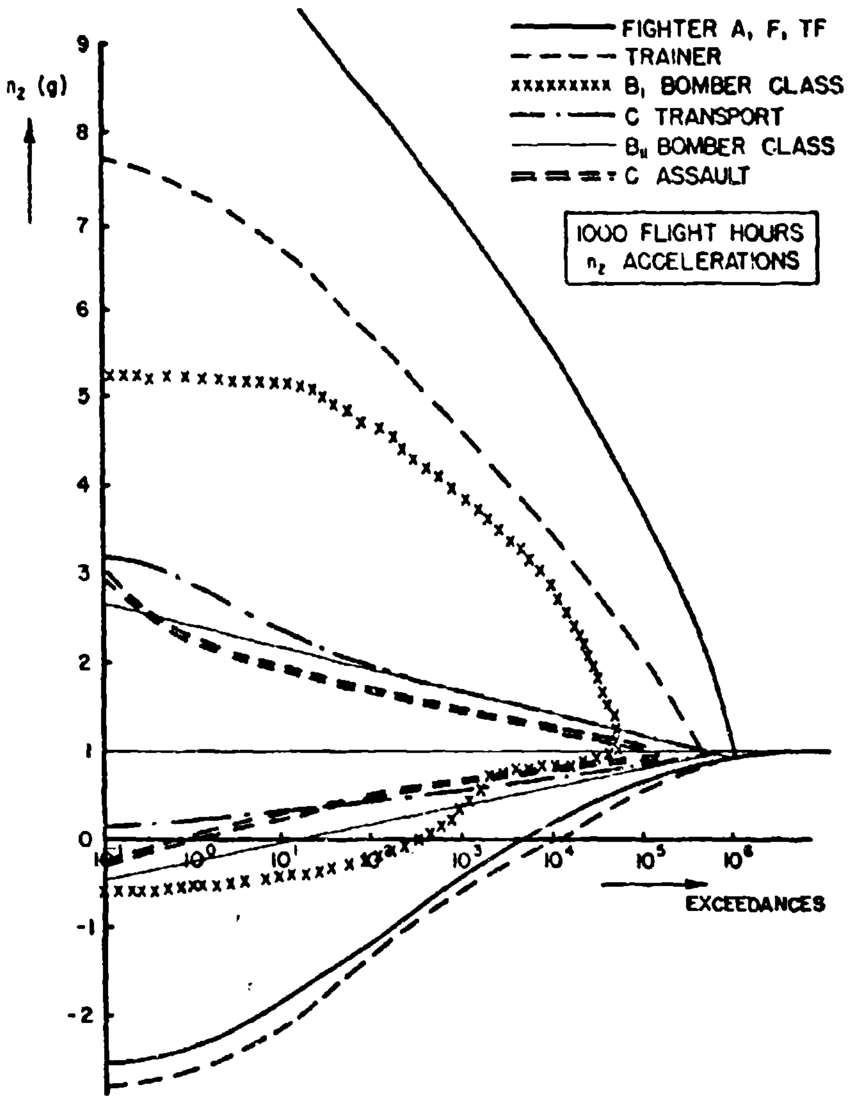


FIGURE 5.30 MANEUVER SPECTRA ACCORDING TO MIL-A-008866B (USAF)

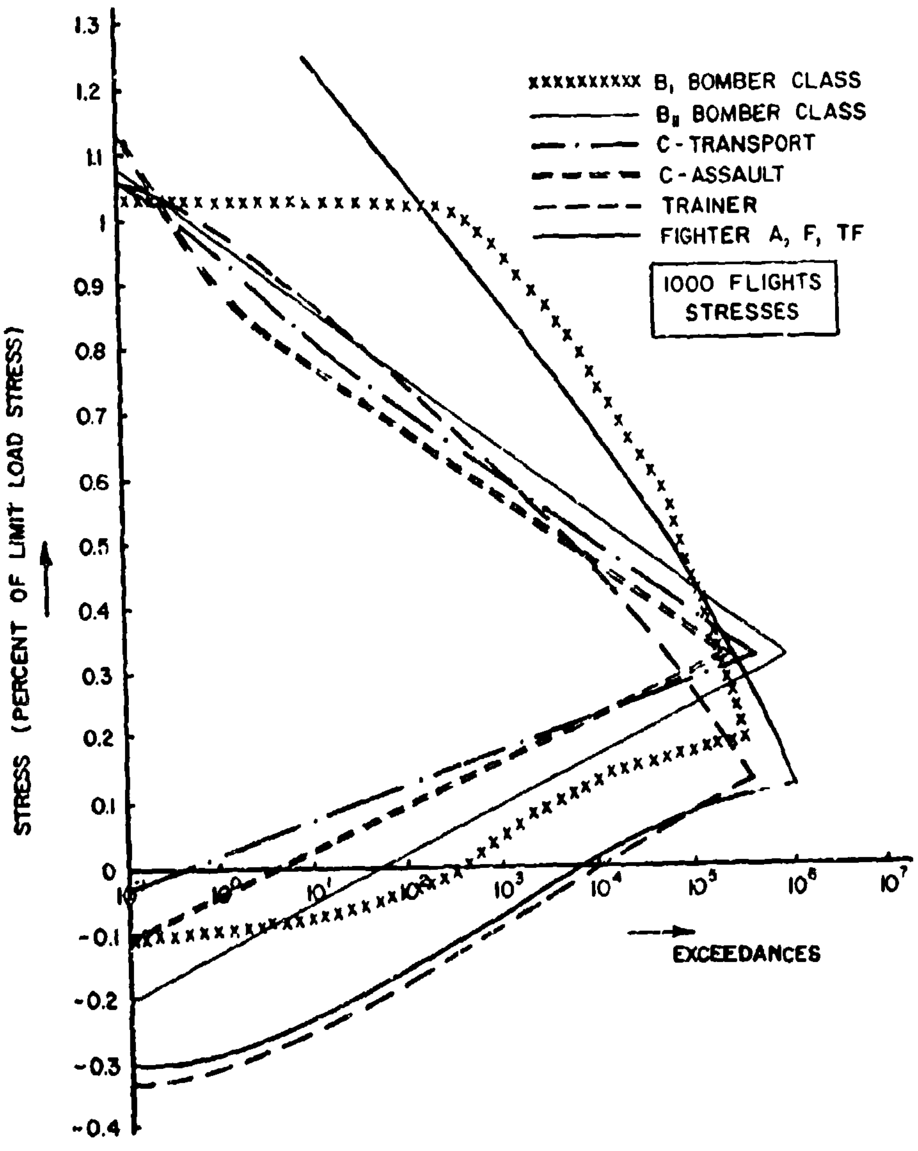
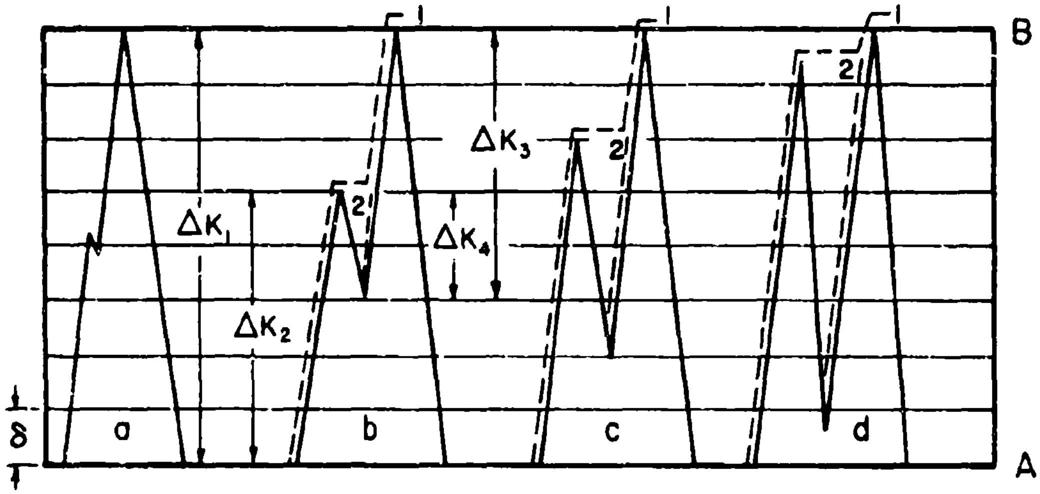


FIGURE 5.31 APPROXIMATE STRESS SPECTRUM FOR 1000 FLIGHTS BASED ON MIL-A-008866B (USAF)



NORMALLY CALCULATED CRACK GROWTH:

a $\Delta a_a = C(\Delta K_1)^4 = C(8\delta)^4 =$ $4096\ C\delta^4$

b $\Delta a_b = C(\Delta K_2)^4 + C(\Delta K_3)^4 = 2C(5\delta)^4 = 1350\ C\delta^4$

c $\Delta a_c =$ $= 2C(6\delta)^4 = 2592\ C\delta^4$

d $\Delta a_d =$ $= 2C(75\delta)^4 = 6338\ C\delta^4$

ALTERNATIVE CRACK GROWTH CALCULATIONS:

a $\Delta a_a = C(\Delta K_1)^4 = C(8\delta)^4 =$ $= 4096\ C\delta^4$

b $\Delta a_b = C(\Delta K_1)^4 + C(\Delta K_4)^4 = C(8\delta)^4 + C(2\delta)^4 = 4112\ C\delta^4$

c $\Delta a_c =$ $= C(8\delta)^4 + C(4\delta)^4 = 4352\ C\delta^4$

d $\Delta a_d =$ $= C(8\delta)^4 + C(7\delta)^4 = 6497\ C\delta^4$

FIGURE 5.32 DEFINITION OF CYCLES

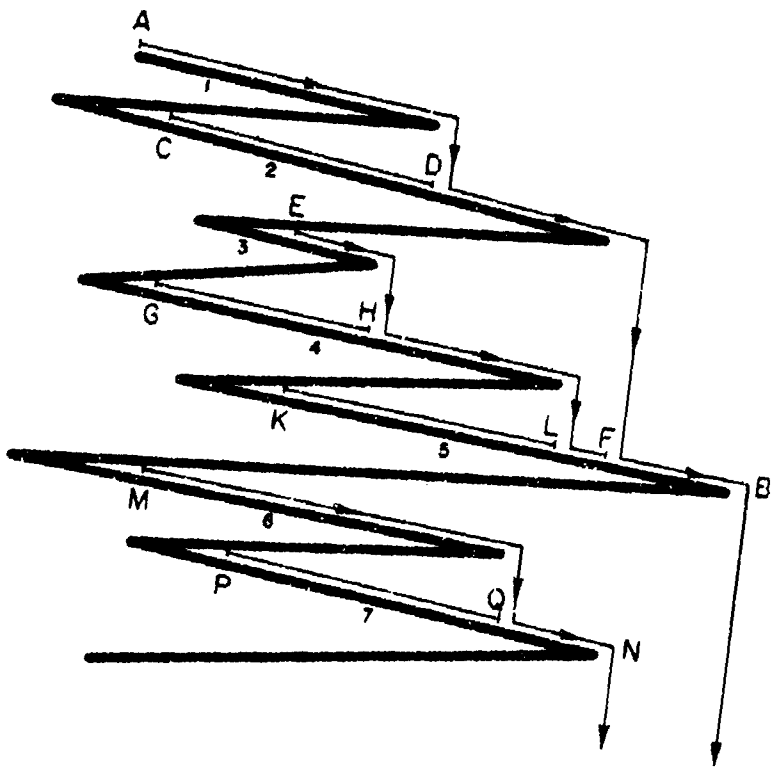


FIGURE 5.33 RAIN FLOW COUNT

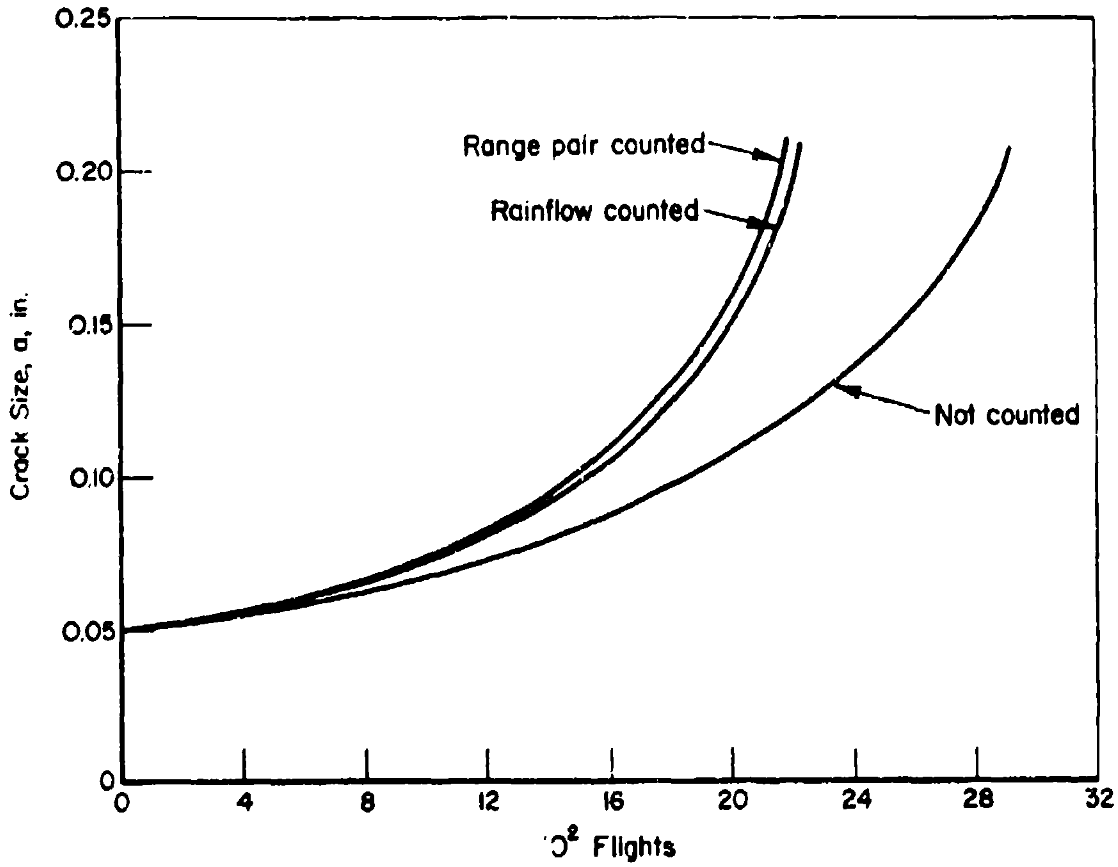


FIGURE 5.34 CALCULATED CRACK GROWTH CURVES FOR RANDOM FLIGHT-BY-FLIGHT FIGHTER SPECTRUM (Ref. 53)

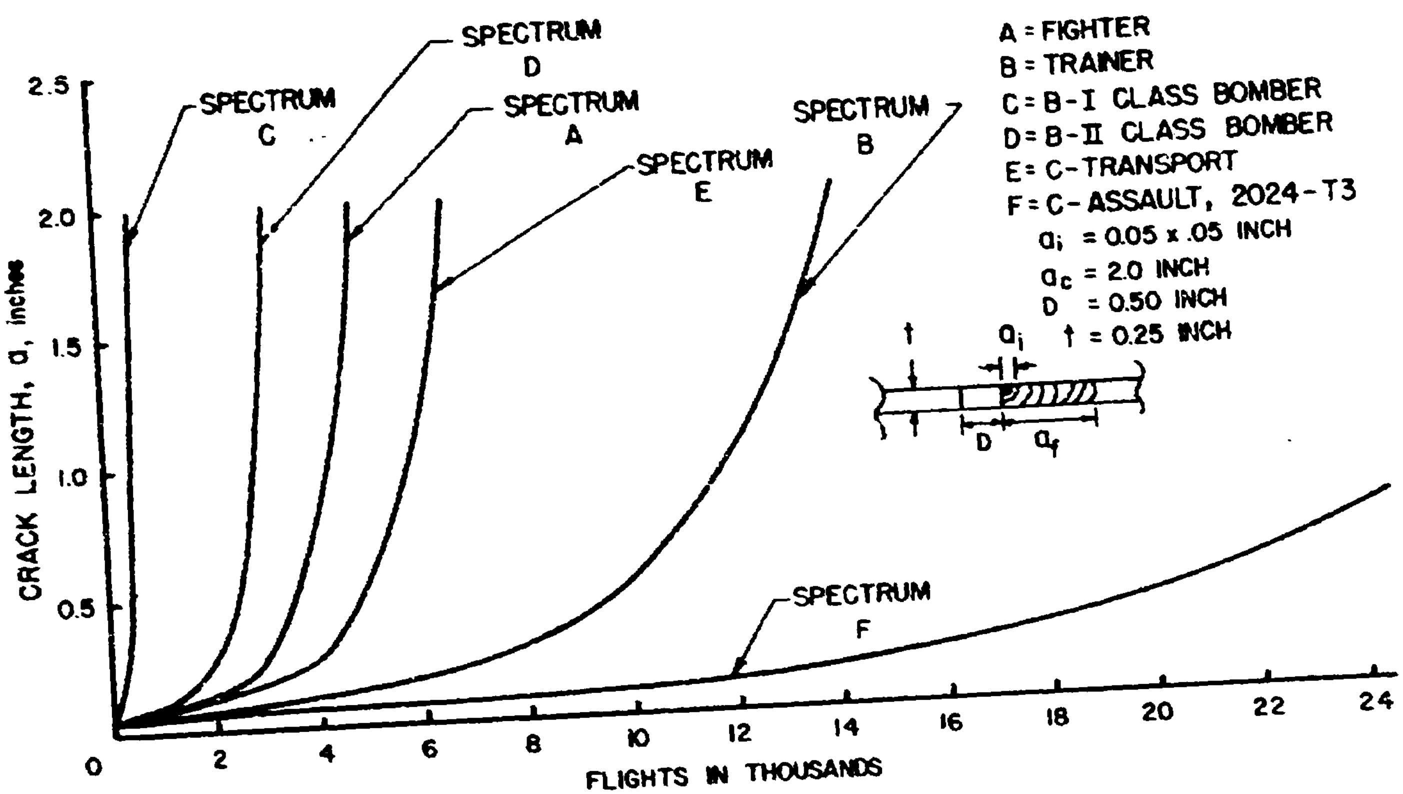


FIGURE 5.35 SPECTRUM FATIGUE CRACK GROWTH BEHAVIOR WILLENBORG RETARDATION CARRY OVER OF RETARDATION

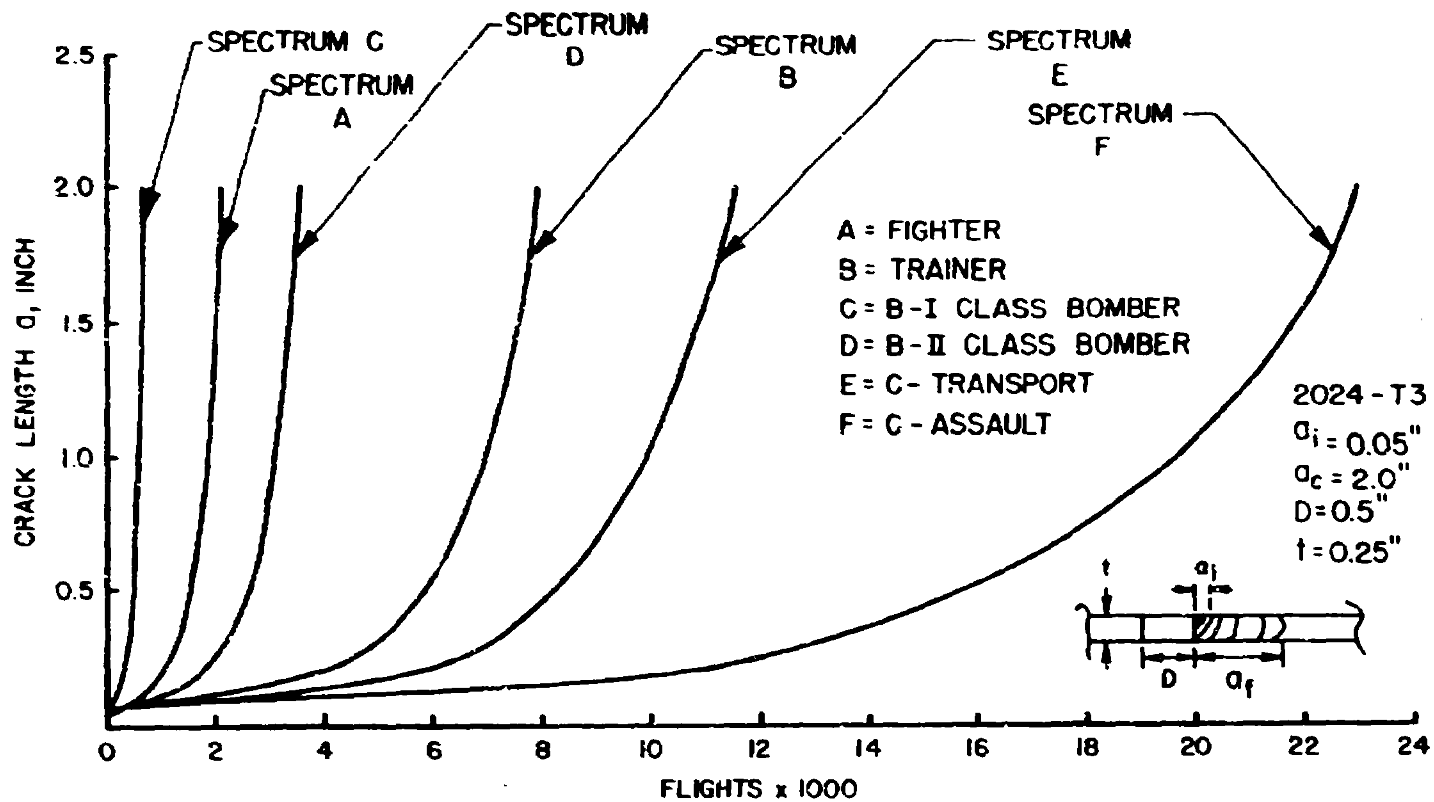


FIGURE 5.36 SPECTRUM FATIGUE CRACK GROWTH BEHAVIOR WHEELER RETARDATION MODEL (CARRY OVER OF RETARDATION)

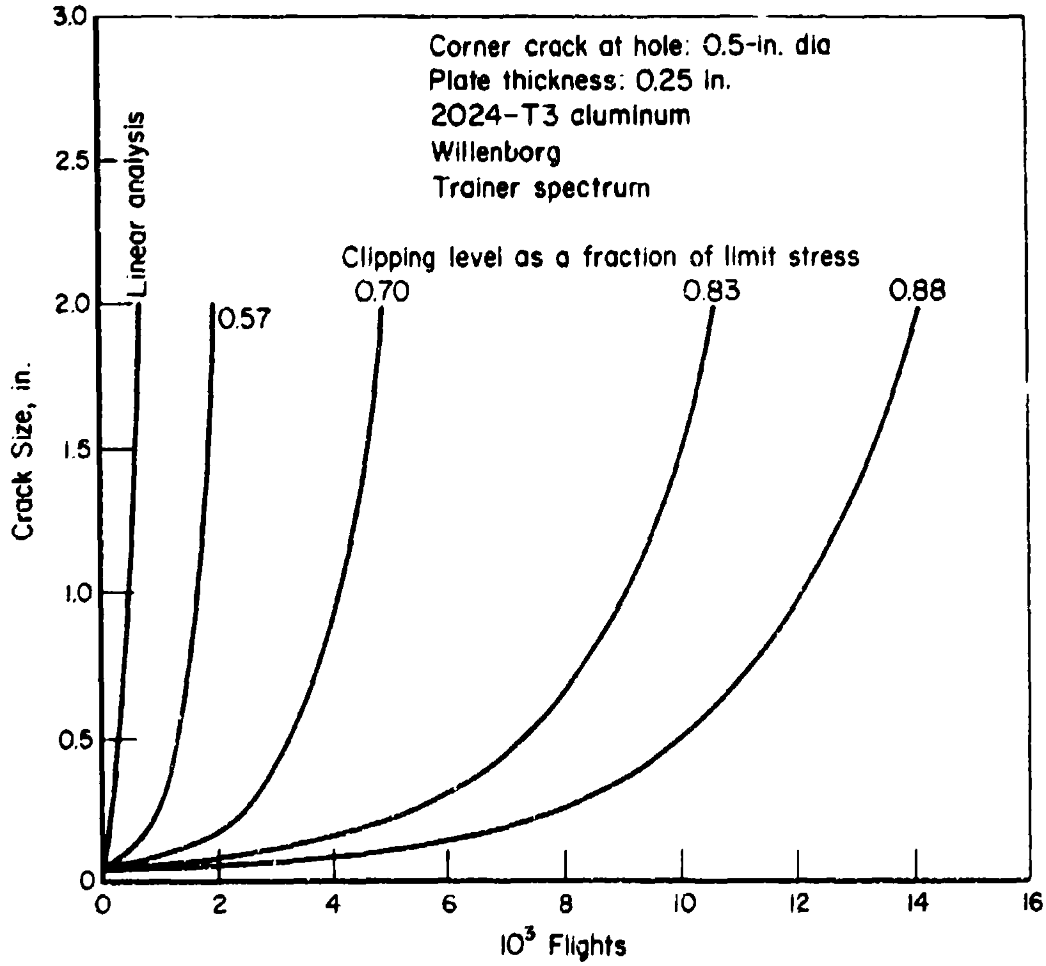


FIGURE 5.37 EFFECT OF CLIPPING LEVEL ON CALCULATED CRACK GROWTH

	SYMBOL	SPECTRUM	LINEAR ANALYSIS (FLIGHTS)	RETARDATION (FLIGHTS)	
				WILLENBORG FULLY RETARDED	m = 2.3 WHEELER
A	▲ WILLENBORG △ WHEELER	FIGHTER	270	4,900	2,100
B	● WILLENBORG ○ WHEELER	TRAINER	460	14,200	7,900
C	■ WILLENBORG □ WHEELER	B-1 CLASS BOMBER	140	700	700
D	▼ WILLENBORG ▽ WHEELER	C-TRANSPORT	1,270	6,700	11,600

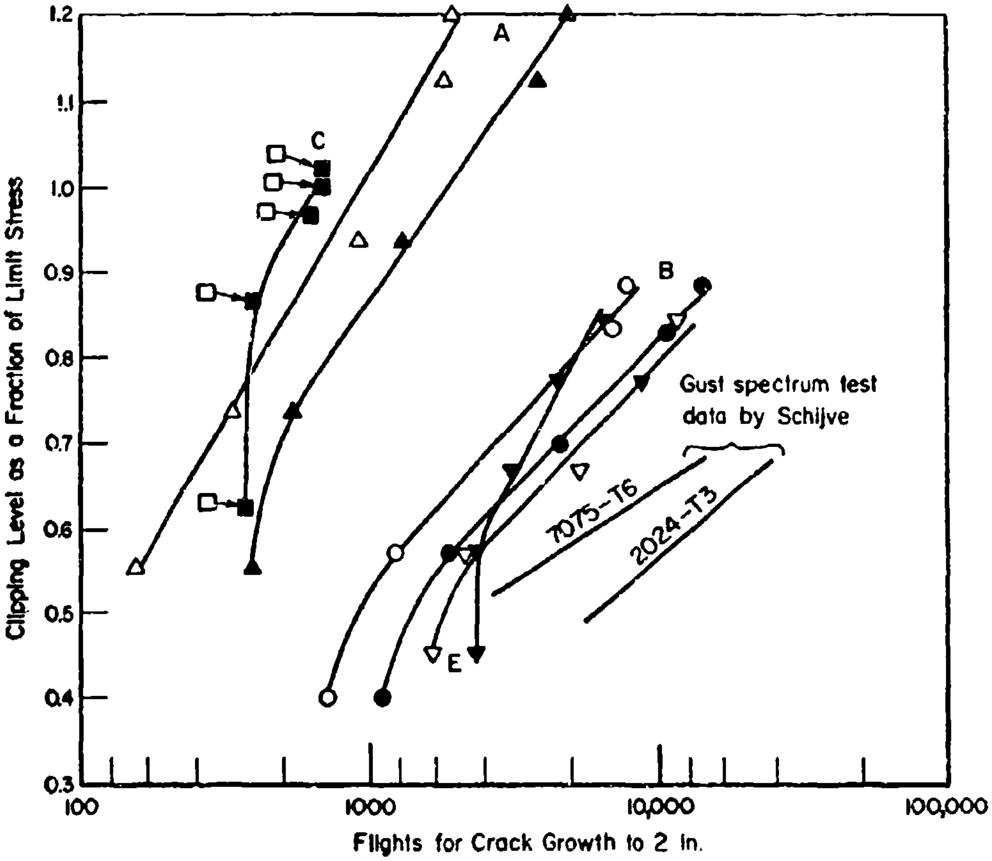


FIGURE 5.38 EFFECT OF CLIPPING FOR VARIOUS SPECTRA

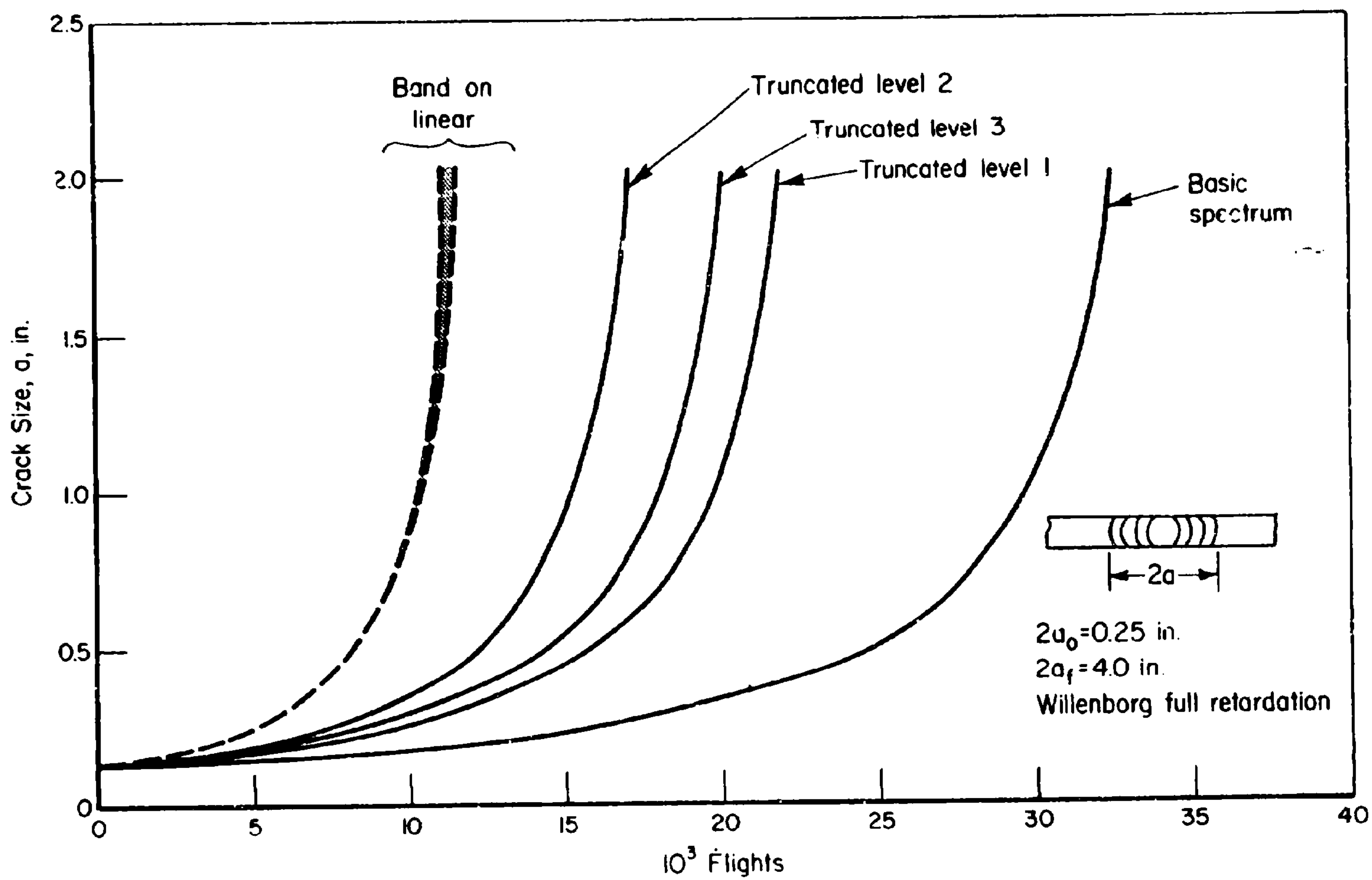


FIGURE 5.39 SPECTRUM CLIPPING EFFECTS ON FATIGUE CRACK GROWTH BEHAVIOR – GUST SPECTRUM

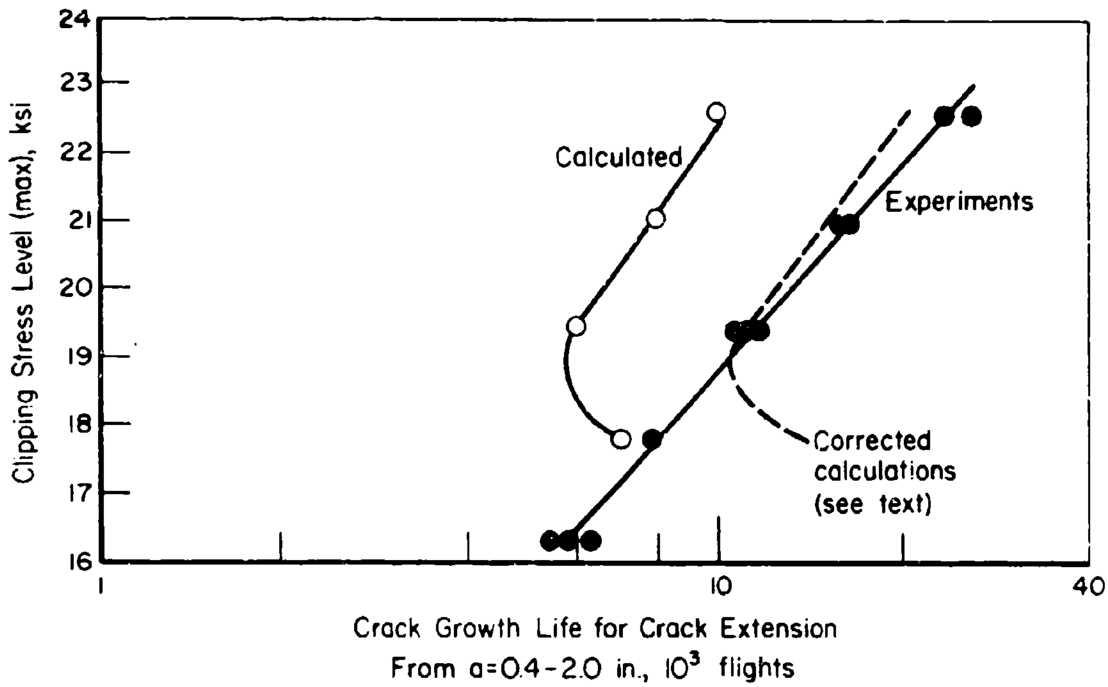


FIGURE 5.40 CALCULATED AND EXPERIMENTAL DATA FOR GUST SPECTRUM CLIPPING (Ref 38, 39)

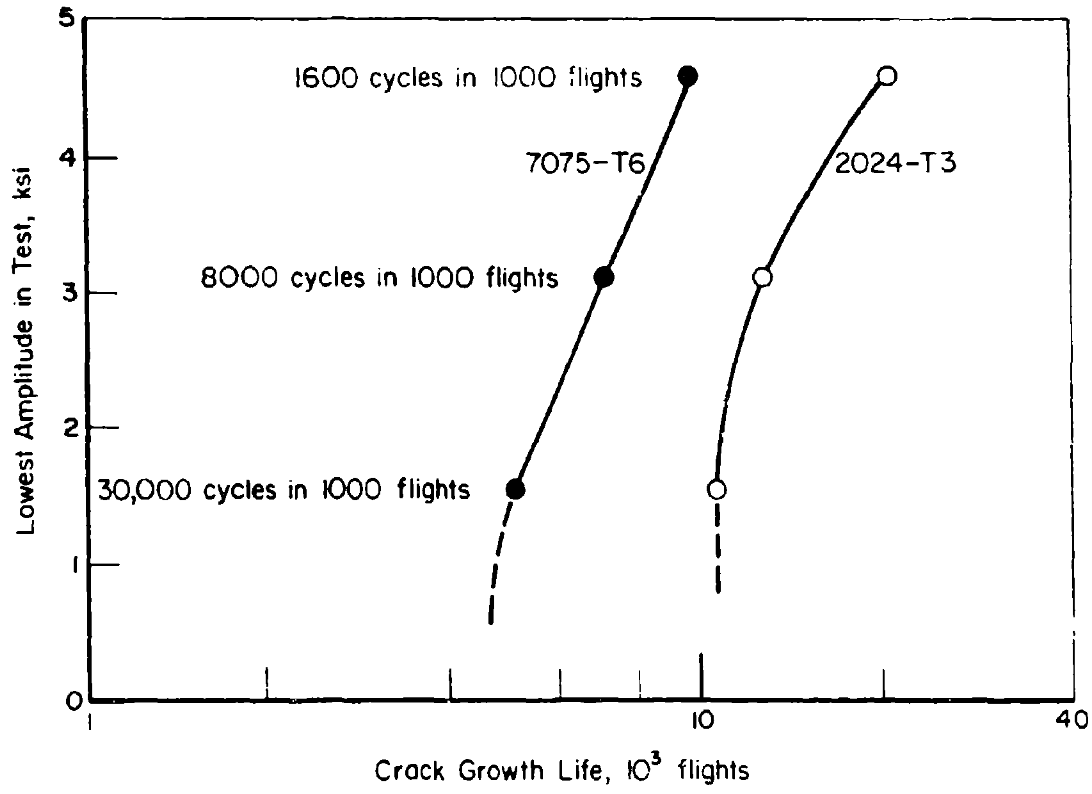
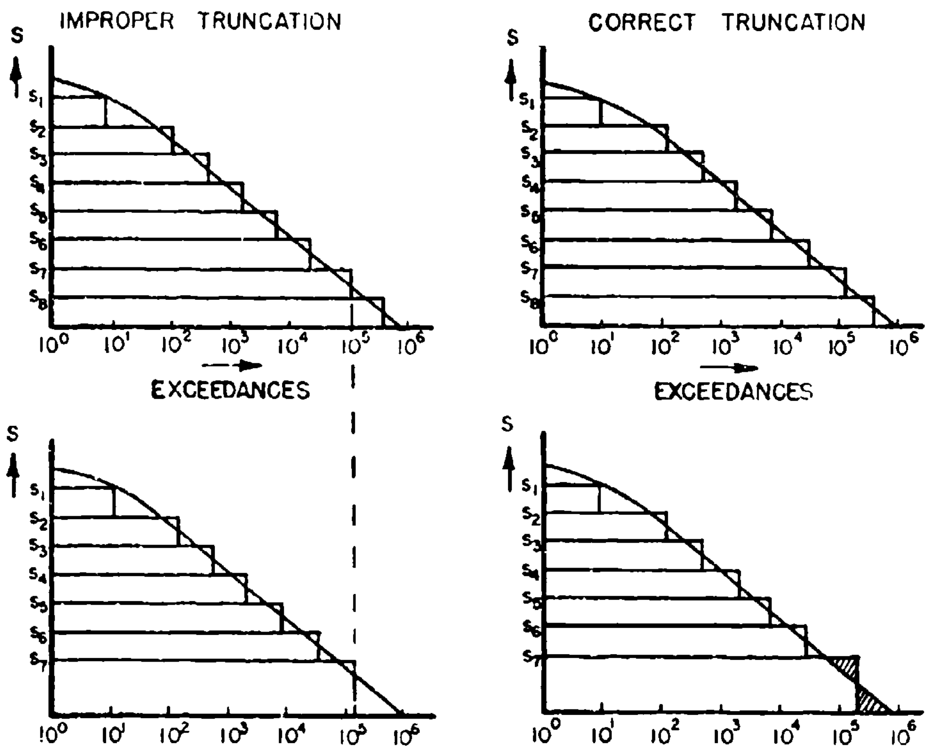


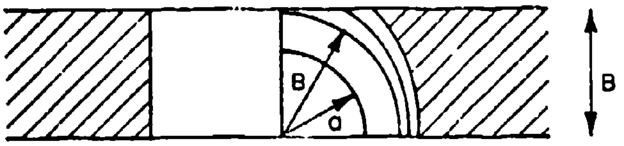
FIGURE 5.41 EFFECT OF LOWEST STRESS AMPLITUDE IN FLIGHT-BY-FLIGHT TESTS BASED ON GUST SPECTRUM (Ref. 38, 39)



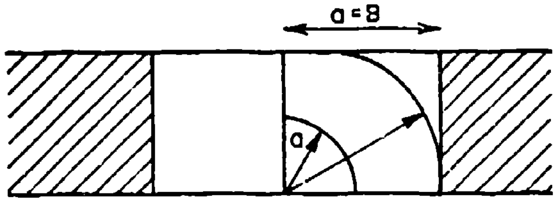
LEVEL	EXCEEDANCES	OCCURRENCES
S_1	10	10
S_2	100	90
S_3	600	500
S_4	2000	1400
S_5	8000	6000
S_6	40000	32000
S_7	120000	80000
S_8	700000	580000

LEVEL	EXCEEDANCES	OCCURRENCES
S_1	10	10
S_2	100	90
S_3	600	500
S_4	2000	1400
S_5	8000	6000
S_6	40000	32000
S_7	300000	260000

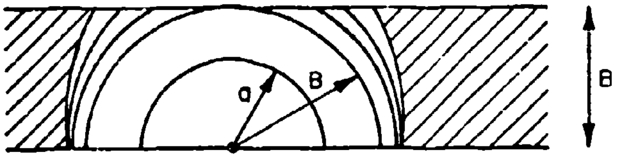
Figure 5.42 Improper and Correct Truncation



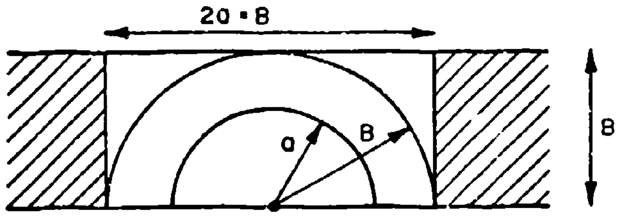
DEVELOPMENT OF CORNER CRACK
INTO THROUGH CRACK



RECOMMENDED FLAW DEVELOPMENT
FOR ANALYSIS PURPOSES



DEVELOPMENT OF SURFACE FLAW
INTO THROUGH CRACK



RECOMMENDED FLAW DEVELOPMENT
FOR ANALYSIS PURPOSES

FIGURE 5.43 DEVELOPMENT OF FLAWS

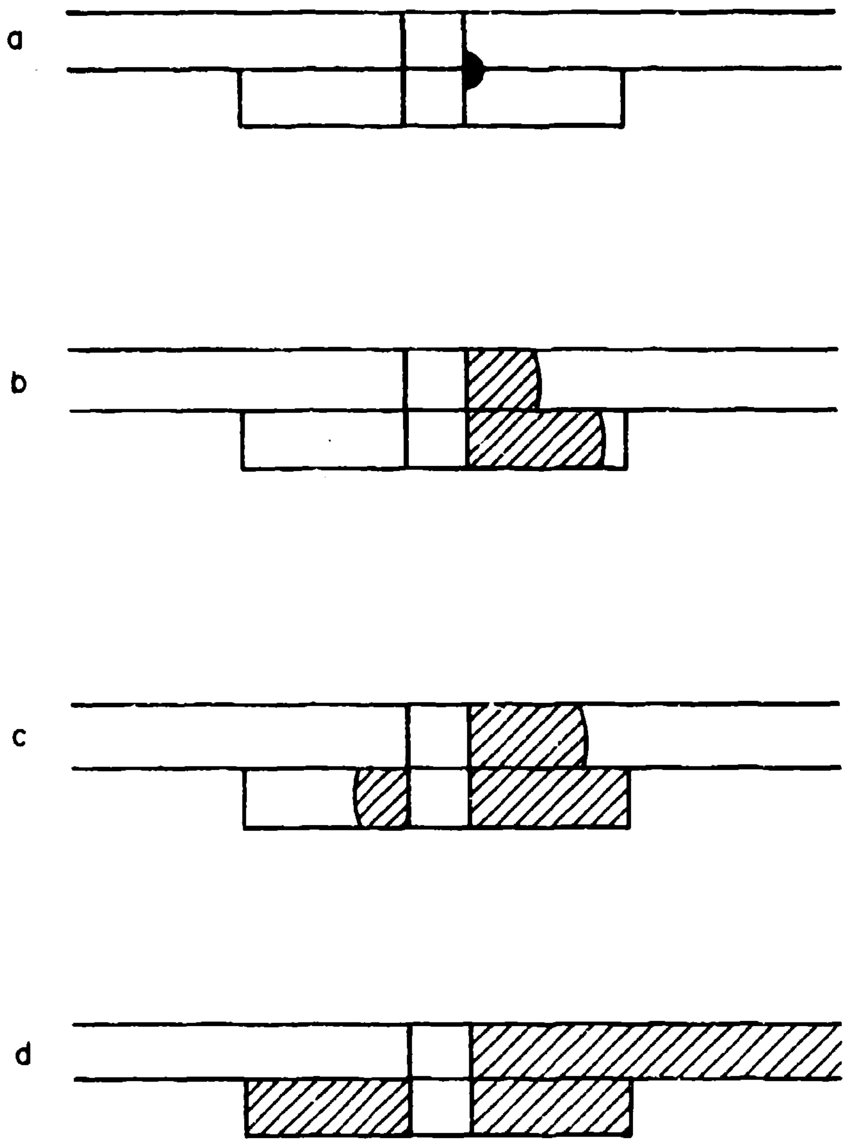


FIGURE 5.44 INTERACTION OF CRACKS

CHAPTER 6

Damage Tolerance Analysis
Sample Problems
(to be added later)

CHAPTER 7

Damage Tolerance Testing
(to be added later)

CHAPTER 8

Individual Aircraft Tracking
(to be added later)

CHAPTER 9

Fracture Control Guidelines
(to be added later)

CHAPTER 10

Design Guidelines
(to be added later)

CHAPTER 11

Repair Guidelines
(to be added later)

APPENDIX:

Specifications and Standards

MIL-STD-1530A(11)
11 December 1975
SUPERSEDING
MIL-STD-1530(USAF)
1 September 1972

MILITARY STANDARD

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM, AIRPLANE REQUIREMENTS

FSC 15GP

Airplane Structural Integrity Program, Airplane Requirements

MIL-STD-1530A(11)

1. This Military Standard is approved for use by all Departments and Agencies of the Department of Defense.

2. Recommended corrections, additions, or deletions should be addressed to Aeronautical Systems Division, ASD/ENFS, Wright-Patterson Air Force Base, Ohio 45433.

CONTENTS

CONTENTS (CONT'D)

CONTENTS (CONT'D)

FIGURES

1. SCOPE

1.1 Purpose. The purpose of this standard is to describe the Air Force Aircraft Structural Integrity Program, define the overall requirements necessary to achieve structural integrity of USAF airplanes, and specify acceptance methods of contractor compliance. This standard shall be used by:

a. Contractors in conducting the development of an airframe for a particular weapon or support system

b. Government personnel in managing the development, production, and operational support of a particular airplane system throughout its life cycle.

1.2 Applicability. The degree of applicability of the various portions of this standard may vary between airplane systems as specified in 1.3.

1.2.1 Type of aircraft. This standard is directly applicable to manned power driven aircraft having fixed or adjustable fixed wings and to those portions of manned helicopter and V/STOL aircraft which have similar structural characteristics. Helicopter-type power transmission systems, including lifting and control rotors, and other dynamic machinery, and power generators, engines, and propulsion systems are not covered by this standard. For unmanned vehicles, certain requirements of this standard may be waived or factors of safety reduced commensurate with sufficient structural safety and durability to meet the intended use of the airframe. Waivers and deviations shall be specified in the contract specifications and shall have specific Air Force approval prior to commitment in the design.

1.2.2 Type of program. This standard applies to.

a. Future airplane systems

b. Airplane systems procured by the Air Force but developed under the auspices of another regulatory activity (such as the FAA or USN)

c. Airplanes modified or directed to new missions.

1.2.3 Type of structure. This standard applies to metallic and nonmetallic structures unless stated otherwise in the specifications referenced herein.

1.3 Modifications. The Air Force will make the decision regarding application of this standard and may modify requirements of this standard to suit system needs. The description of the modifications shall be documented in accordance with 5.1.1.

2. REFERENCED DOCUMENTS

2.1 Issues of documents. The following documents, of the issue in effect on date of invitation for bids or request for proposal, form a part of this standard to the extent specified herein:

SPECIFICATIONS

Military

MIL-I-6870	Inspection Program Requirements, Nondestructive, for Aircraft and Missile Materials and Parts
MIL-A-8860	Airplane Strength and Rigidity, General Specification for
MIL-A-8861	Airplane Strength and Rigidity, Flight Loads
MIL-A-8862	Airplane Strength and Rigidity, Landplane, Landing and Ground Handling Loads
MIL-A-8865	Airplane Strength and Rigidity, Miscellaneous Loads
MIL-A-8866	Airplane Strength and Rigidity, Reliability Requirements, Repeated Loads, and Fatigue
MIL-A-8867	Airplane Strength and Rigidity, Ground Tests
MIL-A-8869	Airplane Strength and Rigidity, Nuclear Weapons Effects
MIL-A-8870	Airplane Strength and Rigidity, Vibration Flutter and Divergence
MIL-A-8871	Airplane Strength and Rigidity, Flight and Ground Operations Tests
MIL-A-8892	Airplane Strength and Rigidity, Vibration
MIL-A-8893	Airplane Strength and Rigidity, Sonic Fatigue
MIL-R-83165	Recorder, Signal Data, MXU-553/A
MIL-C-83166	Converter-multiplexer, Signal Data, General Specification for
MIL-A-83444	Airplane Damage Tolerance Requirements

STANDARDS

Military

MIL-STD-499	Engineering Management
MIL-STD-882	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-1515	Fasteners to be Used in the Design and Construction of Aerospace Mechanical Systems
MIL-STD-1568	Materials and Processes for Corrosion and Prevention and Control in Aerospace Weapons Systems

HANDBOOKS

Military

MIL-HDBK-5	Metallic Materials and Elements for Aerospace Vehicle Structures
MIL-HDBK-17	Plastics for Flight Vehicles
MIL-HDBK-23	Structural Sandwich Composites

Air Force Systems Command Design Handbooks

DH 1-0	General
DH 1-2	General Design Factors
DH 2-0	Aeronautical Systems
DH 2-7	System Survivability

(Copies of specifications, standards, drawings and publications required by contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

2.2 Other publications. The following document forms a part of this standard to the extent specified herein. Unless otherwise indicated, the issue in effect on date of invitation for bids or request for proposal shall apply.

Other Publications

MCIC-HB-01	Damage Tolerance Design Handbook

(Application for copies should be addressed to the Metals and Ceramics Information Center, Battelle Memorial Institute, Columbus, Ohio 43201.)

3. DEFINITIONS. Definitions will be in accordance with the documents listed in Section 2 and as specified herein.

3.1 Durability. The ability of the airframe to resist cracking (including stress corrosion and hydrogen induced cracking), corrosion, thermal degradation, delamination, wear, and the effects of foreign object damage for a specified period of time.

3.2 Economic life. That operational life indicated by the results of the durability test program, i.e., test performance interpretation and evaluation in accordance with MIL-A-8867 to be available with the incorporation of Air Force approved and committed production or retrofit changes and supporting application of the force structural maintenance plan in accordance with this standard. In general, production or retrofit changes will be incorporated to correct local design and manufacturing deficiencies disclosed by test. It will be assumed that the economic life of the test article has been attained with the occurrence of widespread damage which is uneconomical to repair and, if not repaired, could cause functional problems affecting operational readiness. This can generally be characterized by a rapid increase in the number of damage locations or repair costs as a function of cyclic test time.

3.3 Initial quality. A measure of the condition of the airframe relative to flaws, defects, or other discrepancies in the basic materials or introduced during manufacture of the airframe.

3.4 Structural operating mechanisms. Those operating, articulating, and control mechanisms which transmit structural forces during actuation and movement of structural surfaces and elements.

3.5 Damage tolerance. The ability of the airframe to resist failure due to the presence of flaws, cracks, or other damage for a specified period of unrepaired usage.

4. GENERAL REQUIREMENTS

4.1 Discussion. The effectiveness of any military force depends in part on the operational readiness of weapon systems. One major item of an airplane system affecting its operational readiness is the condition of the structure. The complete structure, herein referred to as the airframe, includes the fuselage, wing, empennage, landing gear, control systems and surfaces, engine mounts, structural operating mechanisms, and other components as specified in the contract specification. To maintain operational readiness, the capabilities, condition, and operational limitations of the airframe of each airplane weapon and support system must be established. Potential structural or material problems must be identified early in the life cycle to minimize their impact on the operational force, and a preventive maintenance program must be determined to provide for the orderly scheduling of inspections and replacement or repair of life-limited elements of the airframe.

4.1.1 The overall program to provide USAF airplanes with the required structural characteristics is referred to as the Aircraft Structural Integrity Program (ASIP). General requirements of the ASIP are to:

a. Establish, evaluate, and substantiate the structural integrity (airframe strength, rigidity, damage tolerance, and durability) of the airplane.

b. Acquire, evaluate, and utilize operational usage data to provide a continual assessment of the in-service integrity of individual airplanes.

c. Provide a basis for determining logistics and force planning requirements (maintenance, inspections, supplies, rotation of airplanes, system phaseout, and future force structure).

d. Provide a basis to improve structural criteria and methods of design, evaluation, and substantiation for future airplanes.

4.1.2 The majority of detail requirements are published in the referenced military specifications. This standard repeats some of these requirements for emphasis and contains additional requirements which are not currently included in the military specifications. Any differences in detail requirements that may exist between this standard and the referenced documents listed in Section 2 shall be brought to the immediate attention of the Air
[illegible] for resolution. The applicable specifications, including the latest
[illegible] thereto, for a particular airplane shall be as stated in the con-
[illegible]cifications.

4.2 Requirements. ASIP consists of the following five interrelated functional tasks as specified in table 1 and figures 1, 2, and 3:

a. Task I (design information): Development of those criteria which must be applied during design so that the specific requirements will be met.

b. Task II (design analysis and development tests): Development of the design environment in which the airframe must operate and the response of the airframe to the design environment.

c. Task III (full scale testing): Flight and laboratory tests of the airframe to assist in determination of the structural adequacy of the design.

d. Task IV (force management data package): Generation of data required to manage force operations in terms of inspections, modifications, and damage assessments.

e. Task V (force management): Those operations that must be conducted by the Air Force during force operations to ensure damage tolerance and durability throughout the useful life of individual airplanes.

5. DETAIL REQUIREMENTS

5.1 Design information (Task I). The design information task encompasses those efforts required to apply the existing theoretical, experimental, applied research, and operational experience to specific criteria for materials selection and structural design for the airplane. The objective is to ensure that the appropriate criteria and planned usage are applied to an airplane design so that the specific operational requirements will be met. This task begins as early as possible in the conceptual phase and is finalized in subsequent phases of the airplane life cycle.

5.1.1 ASIP master plan. The contractor shall prepare an ASIP Master Plan in accordance with the detail requirements specified in the contract specifications. The purpose of the ASIP Master Plan is to define and document the specific approach for accomplishment of the various ASIP tasks throughout the life cycle of the airplane. The plan shall depict the time phased scheduling and integration of all required ASIP tasks for design, development, qualification, and tracking of the airframe. The plan shall include discussion of unique features, exceptions to the requirements of this standard and the associated rationale, and any problems anticipated in the execution of the plan. The development of the schedule shall consider all interfaces, impact of schedule delays (e.g., delays due to test failure), mechanisms for recovery programming, and other problem areas. The plan and schedules shall be updated annually and when significant changes occur. The ASIP Master Plan shall be subject to approval by the Air Force.

5.1.2 Structural design criteria. Detail structural design criteria for the specific airplane shall be established by the contractor in accordance with the requirements of the specifications as specified in 5.1.2.2. These specifications contain design criteria for strength, damage tolerance, durability, flutter, vibration, sonic fatigue, and weapons effects. The structural design criteria for damage tolerance and durability are further specified in 5.1.2.1 for special emphasis.

5.1.2.1 Damage tolerance and durability design criteria. The airframe shall incorporate materials, stress levels, and structural configurations which:

a. Allow routine in-service inspection

b. Minimize the probability of loss of the airplane due to propagation of undetected cracks, flaws, or other damage

c. Minimize cracking (including stress corrosion and hydrogen induced cracking), corrosion, delamination, wear, and the effects of foreign object damage.

Damage tolerance design approaches shall be used to insure structural safety since undetected flaws or damage can exist in critical structural components despite the design, fabrication, and inspection efforts expended to eliminate their occurrence. Durability structural design approaches shall be used to achieve Air Force weapon and support systems with low in-service maintenance costs and improved operational readiness throughout the design service life of the airplane.

5.1.2.1.1 Damage tolerance. The damage tolerance design requirements are specified in MIL-A-83444, and shall apply to safety-of-flight structure. Damage tolerance designs are categorized into two general concepts:

a. Fail-safe concepts where unstable crack propagation is locally contained through the use of multiple load paths or tear stoppers

b. Slow crack growth concepts where flaws or defects are not allowed to attain the size required for unstable rapid propagation.

Either design concept shall assume the presence of undetected flaws or damage, and shall have a specified residual strength level both during and at the end of a specified period of unrepaired service usage. The initial damage size assumptions, damage growth limits, residual strength requirements and the minimum periods of unrepaired service usage depend on the type of structure and the appropriate inspectability level.

5.1.2.1.2 Durability. The durability design requirements are specified in MIL-A-8866. The airframe shall be designed such that the economic life is greater than the design service life when subjected to the design service loads/environment spectrum. The design service life and typical design usage requirements will be specified by the Air Force in the contract specifications for each new airplane. The design objective is to minimize cracking or other structural or materi degradation which could result in excessive maintenance problems or functional problems such as fuel leakage, loss of control effectiveness, or loss of cabin pressure.

5.1.2.2 Structural design criteria requirements. Using the requirements in the System specification and the referenced military specifications the contractor shall prepare the detailed structural design criteria for the particular airplane. These criteria and all elements thereof shall require approval by the Air Force. Detail structural design criteria are specified in AFSC DH 1-0 and DH 2-0 and in MIL-A-8860, MIL-A-8861, MIL-A-8862, MIL-A-8865, MIL-A-8866, MIL-A-8869, MIL-A-8870, MIL-A-8892, MIL-A-8893, and MIL-A-83444. Where applicable, specific battle damage criteria will be provided by the Air Force. These criteria will include the threat, flight conditions, and load carrying capability and duration after damage is imposed, etc. The structure shall be designed to these criteria and to other criteria as specified in AFSC DH 2-7.

5.1.3 Damage tolerance and durability control plans. The contractor shall prepare damage tolerance and durability control plans and conduct the resulting programs in accordance with this standard, MIL-A-8866, and MIL-A-83444. The plans shall identify and define all of the tasks necessary to ensure compliance with the damage tolerance requirements as specified in 5.1.2.1.1 and MIL-A-83444, and the durability requirements as specified in 5.1.2.1.2 and MIL-A-8866. The plans and their individual elements shall require approval by the Air Force. The disciplines of fracture mechanics, fatigue, materials selection and processes, environmental protection, corrosion prevention and control, design, manufacturing, quality control, and nondestructive inspection are involved in damage tolerance and durability control. The corrosion prevention and control plan shall be in accordance with MIL-STD-1568. The plans shall include the requirement to perform damage tolerance and durability design concepts/material/weight/performance/cost trade studies during the early design phases to obtain low weight, cost effective designs which comply with the requirements of MIL-A-8866 and MIL-A-83444.

5.1.3.1 Damage tolerance control plan. The damage tolerance control plan shall include as a minimum the following tasks:

a. Basic fracture data (i.e., K_{IC}, K_C, K_{ISCC}, da/dn, etc.) utilized in the initial trade studies and the final design and analyses shall be obtained from existing sources or developed as part of the contract in accordance with 5.2.1.

b. A fracture critical parts list shall be established by the contractor in accordance with MIL-A-83444. The fracture critical parts list shall require approval by the Air Force and the list shall be kept current as the design of the airframe progresses.

c. Design drawings for the fracture critical parts shall identify critical locations and special processing (e.g., shot peening) and inspection requirements.

d. Complete nondestructive inspection requirements, process control requirements, and quality control requirements for fracture critical parts shall be established by the contractor and shall require approval by the Air Force. Nondestructive inspections shall comply with MIL-I-6870. This task shall include the proposed plan for certifying and monitoring subcontractor, vendor, and supplier controls.

e. The damage tolerance control plan shall include any special nondestructive inspection demonstration programs conducted in accordance with the requirements of MIL-A-83444.

f. Material procurement and manufacturing process specifications shall be developed and updated as necessary to minimize the possibility that basic materials and the resulting fracture critical parts have fracture toughness properties in the important loading directions which are less than those used in design.

g. Traceability requirements shall be defined and imposed by the contractor on those fracture critical parts that receive prime contractor or subcontractor in-house processing and fabrication operations which could degrade the design material properties.

h. Damage tolerance analyses, development testing, and full scale testing shall be performed in accordance with this standard, MIL-A-8867 and MIL-A-83444.

i. For all fracture critical parts that are designed for a degree of inspectability other than in-service noninspectable, the contractor shall define the necessary inspection procedures for field use for each appropriate degree of inspectability as specified in MIL-A-83444.

5.1.3.2 Durability control plan. The durability control plan shall include as a minimum the following tasks:

a. A disciplined procedure for durability design shall be implemented to minimize the possibility of incorporating adverse residual stresses, local design details, materials, processing, and fabrication practices into the airplane design and manufacture which could lead to cracking or failure problems (i.e., those problems which have historically been found early during durability testing or early in service usage). The durability control plan shall encompass the requirements specified in the durability detail design procedures of MIL-A-8866.

b. Basic data (i.e., initial quality distribution, fatigue allowables, etc.) utilized in the initial trade studies and the final design and analyses shall be obtained from existing sources or developed as part of the contract in accordance with 5.2.1.

c. A criteria for identifying durability critical parts shall be established by the contractor and shall require approval by the Air Force. It is envisioned that durability critical parts will be expensive, noneconomical-to-replace parts that are either designed and sized by the durability requirements of MIL-A-8866 or could be designed and sized by the requirements of MIL-A-8866 if special control procedures are not employed. A durability critical parts list shall be established by the contractor and shall be kept current as the design of the airframe progresses.

d. Design drawings for the durability critical parts shall identify critical locations and special processing and inspection requirements.

e. Material procurement and manufacturing process specifications shall be developed and updated as necessary to minimize the possibility that initial quality is degraded below that assumed in the design.

f. Experimental determination sufficient to estimate initial quality by microscopic or fractographic examination shall be required for those structural areas where cracks occur during full scale durability testing. The findings shall be used in the full scale test data interpretation and evaluation task as specified in 5.3.8 and, as appropriate, in the development of the force structural maintenance plan as specified in 5.4.3.

g. Durability analyses, development testing, and full scale testing shall be performed in accordance with this standard, MIL-A-8866, and MIL-A-8867.

5.1.4 Selection of materials, processes, and joining methods. Materials, processes, and joining methods shall be selected to result in a light-weight, cost-effective airframe that meets the strength, damage tolerance, and durability requirements of this standard and supporting specifications. A primary factor in the final selection shall be the results of the design concept/material/weight/cost trade studies performed as a part of the damage tolerance and durability control programs.

5.1.4.1 Structural materials, processes, and joining methods selection requirements. In response to the request for proposal, prospective contractors shall identify the proposed materials, processes, and joining methods to be used in each of the structural components and the rationale for the individual selections. After contract award and during the design activity, the contractor shall document the complete rationale used in the final selection for each structural component. This rationale shall include all pertinent data upon which the selections were based including the data base, previous experience, and trade study results. The requirements of AFSC DH 1-2, Sections 7A, paragraph entitled, Materials, and 7B, paragraph entitled, Processes, shall be met as applicable. The selection of fasteners shall be in accordance with MIL-STD-1515. The materials, processes, and joining method selections for fracture and durability critical parts shall require approval by the Air Force.

5.1.5 Design service life and design usage. The Air Force will provide the required design service life and typical design usage as part of the contract specifications. These data shall be used in the initial design and analysis of the airframe. The design service life and design usage will be established through close coordination between the procuring activity and the advanced planning activities (i.e., Hq USAF, Hq AFSC, Hq AFLC, and using commands). Design mission profiles and mission mixes which are realistic estimates of expected service usage will be established. It is recognized that special force management actions will probably be required (i.e., early retirement, early modification, or rotation of selected airplanes) if the actual usage is more severe than the design usage. All revisions in these data subsequent to contract negotiations shall be at the discretion of the Air Force but will require separate negotiations between the Air Force and contractor.

5.2 Design analyses and development tests (Task II). The objectives of the design analyses and development tests task are to determine the environments in which the airframe must operate (load, temperature, chemical, abrasive, vibratory and acoustic environment) and to perform preliminary analyses and tests based on these environments to design and size the airframe to meet the required strength, damage tolerance, and durability requirements.

5.2.1 Material and joint allowables. The contractor shall utilize as appropriate the materials and joint allowables data in MIL-HDBK-5, MIL-HDBK-17, MIL-HDBK-23, and MCIC-HDBK-01 to support the various design analyses. Other data sources may also be used but will require approval by the Air Force.

For those cases where there are insufficient data available, the contractor shall formulate and perform experimental programs to obtain the data. Generation and analysis of test data shall meet the requirements of MIL-HDBK-5. The scope of these programs shall be defined by the prospective contractors in their responses to the request for proposal and shall require approval by the Air Force.

5.2.2 Loads analysis. The contractor shall comply with the detail requirements for loads analysis as specified in the contract specifications. The loads analysis shall consist of determining the magnitude and distribution of significant static and dynamic loads which the airframe may encounter when operating within the envelope established by the structural design criteria. This analysis consists of determining the flight loads, ground loads, powerplant loads, control system loads, and weapon effects. When applicable, this analysis shall include the effects of temperature, aeroelasticity, and dynamic response of the airframe.

5.2.3 Design service loads spectra. The contractor shall comply with the detail requirements for design service loads spectra in MIL-A-8866 as specified in the contract specifications. These spectra shall require approval by the Air Force. The purpose of the design service loads spectra is to develop the distribution and frequency of loading that the airframe will experience based on the design service life and typical design usage. The design service loads spectra and the design chemical/thermal environment spectra as specified in 5.2.4 will be used to develop design flight-by-flight stress/environment spectra as appropriate to support the various analyses and test tasks specified herein.

5.2.4 Design chemical/thermal environment spectra. The contractor shall comply with the detail requirements for design chemical/thermal environment spectra in MIL-A-8866 as specified in the contract specifications. These spectra shall require approval by the Air Force. These spectra shall characterize each environment (i.e., intensity, duration, frequency of occurrence, etc.).

5.2.5 Stress analysis. The contractor shall comply with the detail requirements for stress analysis as specified in the contract specifications. This analysis shall require approval by the Air Force. The stress analysis shall consist of the analytical determination of the stresses, deformation, and margins of safety resulting from the external loads and temperatures imposed on the airframe. The ability of the airframe to support the critical loads and to meet the specified strength requirements shall be established. In addition to verification of strength the stress analysis shall be used as a basis for durability and damage tolerance analyses, selection of critical structural components for design development tests, material review actions, and selection of loading conditions to be used in the structural testing.

The stress analysis shall also be used as a basis to determine the adequacy of structural changes throughout the life of the airplane and to determine the adequacy of the structure for new loading conditions that result from increased performance or new mission requirements. The stress analysis shall be revised to reflect any major changes to the airframe or to the loading conditions applied to the airframe.

5.2.6 Damage tolerance analysis. The contractor shall comply with the detail requirements for damage tolerance analysis in MIL-A-83444 as specified in the contract specifications. This analysis shall require approval by the Air Force. The purpose of this analysis is to substantiate the ability of the structural components to meet the requirements of MIL-A-83444.

5.2.6.1 Analysis procedures. The design flight-by-flight stress/environment spectra based on the requirements of 5.2.3 and 5.2.4 shall be used in the damage growth analysis and verification tests. The calculations of critical flaw sizes, residual strengths, safe crack growth periods, and inspection intervals shall be based on existing fracture test data and basic fracture allowables data generated as a part of the design development test program. The effect of variability in fracture properties on the analytical results shall be accounted for in the damage tolerance design.

5.2.7 Durability analysis. The contractor shall comply with the detail requirements for durability analysis in MIL-A-8866 as specified in the contract specifications. This analysis shall require approval by the Air Force. The purpose of this analysis is to substantiate the ability of the structure to meet the requirements of MIL-A-8866.

5.2.7.1 Analysis procedures. The design flight-by-flight stress/environment spectra based on the requirements of 5.2.3 and 5.2.4 shall be used in the durability analysis and verification tests. The analysis approach shall account for those factors affecting the time for cracks or equivalent damage to reach sizes large enough to cause uneconomical functional problems, repair, modification, or replacement. These factors shall include initial quality and initial quality variations, chemical/thermal environment, load sequence and environment interaction effects, material property variations, and analytical uncertainties. In addition to providing analytical assurance of a durable design, the durability analysis will provide a basis for development of test load spectra to be used in the design development and full scale durability tests.

5.2.8 Sonic durability analysis. The contractor shall comply with the detail requirements for sonic durability analysis in MIL-A-8893 as specified in the contract specifications. This analysis shall require approval by the Air Force. The objective of the sonic durability analysis is to ensure that the airframe is resistant to sonic durability cracking throughout the design service life.

The analysis shall define the intensity of the acoustic environment from potentially critical sources and shall determine the dynamic response, including significant thermal effects. Potentially critical sources include but are not limited to powerplant noise, aerodynamic noise in regions of turbulent and separated flow, exposed cavity resonance, and localized vibratory forces.

5.2.9 Vibration analysis. The contractor shall comply with the detail requirements for vibration analysis in MIL-A-8892 as specified in the contract specifications. This analysis shall require approval by the Air Force. The design shall control the structural vibration environment and the analysis shall predict the resultant environment in terms of vibration levels in various areas of the airplane such as the crew compartment, cargo areas, equipment bays, etc. The structure in each of these areas shall be resistant to unacceptable cracking as specified in 5.2.7.1 due to vibratory loads throughout the design service life. In addition, the design shall control the vibration levels to that necessary for the reliable performance of personnel and equipment throughout the design life of the airplane.

5.2.10 Flutter and divergence analysis. The contractor shall comply with the detail requirements for flutter and divergence analysis in MIL-A-8870 as specified in the contract specifications. This analysis shall require approval by the Air Force. The analysis shall consist of determination of the airplane flutter and divergence characteristics resulting from the interaction of the aerodynamic, inertia, and elastic characteristics of the components involved. The objective of the analysis is to substantiate the ability of the airplane structure to meet the specified flutter and divergence margins. Flutter analysis for failure modes as agreed to by the Air Force and the contractor shall also be conducted.

5.2.11 Nuclear weapons effects analyses. The contractor shall comply with the detail requirements for nuclear weapons effects analyses in MIL-A-8869 as specified in the contract specifications. These analyses shall require approval by the Air Force. The objectives of the nuclear weapons effects analyses are to:

a. Verify that the design of the airframe will successfully resist the specified environmental conditions with no more than the specified residual damage

b. Determine the structural capability envelope and crew radiation protection envelope for other degrees of survivability (damage) as may be required.

The contractor shall prepare detail design criteria and shall conduct the nuclear weapons effects analyses for transient thermal, overpressure, and gust loads and provide the substantiation of allowable structural limits on the structures critical for these conditions. The contractor shall also prepare and report the nuclear weapons effects capability envelope, including crew radiation protection, for a specified range of variations of weapon delivery trajectories, weapon size, aircraft escape maneuvers, and the resulting damage limits.

5.2.12 Non-nuclear weapons effects analysis. The contractor shall comply with the detail requirements for non-nuclear weapons effects analysis in AFSC DH 2-7 as specified in the contract specifications. This analysis shall require approval by the Air Force.

5.2.13 Design development tests. The contractor shall comply with the detail requirements for design development tests in MIL-A-8867, MIL-A-8870, MIL-A-8892, and MIL-A-8893 as specified in the contract specifications. The design development test program shall require approval by the Air Force. The objectives of the design development tests are to establish material and joint allowables; to verify analysis procedures; to obtain early evaluation of allowable stress levels, material selections, fastener systems, and the effect of the design chemical/thermal environment spectra; to establish flutter characteristics through wind tunnel tests; and to obtain early evaluation of the strength, durability (including sonic durability), and damage tolerance of critical structural components and assemblies. Examples of design development tests are tests of coupons; small elements; splices and joints; panels; fittings; control system components and structural operating mechanisms; and major components such as wing carry through, horizontal tail spindles, wing pivots, and assemblies thereof. Prospective contractors shall establish the scope of their proposed test program in their response to the request for proposal. After contract award and during the design analysis task, the contractor(s) shall finalize the plans and submit them to the Air Force for approval. The contractor shall revise and maintain approved updated versions of the test plans as the design develops. The plans shall consist of information such as rationale for selection of scope of tests, description of test articles, procedures, test loads and test duration; and analysis directed at establishing cost and schedule trade-offs used to develop the program.

5.3 Full scale testing (Task III). The objective of this task is to assist in determining the structural adequacy of the basic design through a series of ground and flight tests.

5.3.1 Static tests. The contractor shall comply with the detail requirements for static tests in MIL-A-8867 as specified in the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall be subject to approval by the Air Force. The static test program shall consist of a series of laboratory tests conducted on an instrumented airframe that simulates the loads resulting from critical flight and ground handling conditions. Thermal environment effects shall be simulated along with the load application on airframes where operational environments impose significant thermal effects. The primary purpose of the static test program is to verify the design ultimate strength capabilities of the airframe. Full scale static tests to design ultimate loads shall be required except:

a. Where it is shown that the airframe and its loading are substantially the same as that used on previous aircraft where the airframe has been verified by full scale tests

b. Where the strength margins (particularly for stability critical structure) have been demonstrated by major assembly tests.

When full scale ultimate load static tests are not performed, it shall be a program requirement to conduct a strength demonstration proof test. Deletion of the full scale ultimate load static tests shall require approval by the Air Force. Functional and inspection type proof test requirements shall be in accordance with MIL-A-8867.

5.3.1.1 Schedule requirement. The full scale static tests shall be scheduled such that the tests are completed in sufficient time to allow removal of the 80 percent limit restrictions on the flight test airplanes in accordance with MIL-A-8871 and allow unrestricted flight within the design envelope on schedule.

5.3.2 Durability tests. The contractor shall comply with the detail requirements for durability tests in MIL-A-8867 as specified by the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall require approval by the Air Force. Durability tests of the airframe shall consist of repeated application of the flight-by-flight design service loads/environment spectra. The objectives of the full scale durability tests are to:

a. Demonstrate that the economic life of the test article is equal to or greater than the design service life when subjected to the design service loads/environment spectra

b. Identify critical areas of the airframe not previously identified by analysis or component testing

c. To provide a basis for establishing special inspection and modification requirements for force airplanes.

5.3.2.1 Selection of test articles. The test article shall be an early Full Scale Development (FSD) or Research Development Test & Evaluation (RDT&E) airframe and shall be as representative of the operational configuration as practical. If there are significant design, material, or manufacturing changes between the test article and production airplanes, durability tests of an additional article or selected components and assemblies thereof shall be required.

5.3.2.2 Schedule requirements. The full scale airframe durability test shall be scheduled such that one lifetime of durability testing plus an inspection of critical structural areas in accordance with 5.3.2.2.a and b shall be completed prior to full production go ahead decision. Two lifetimes of durability testing plus an inspection of critical structural areas in accordance with 5.3.2.3.a and b shall be scheduled to be completed prior to delivery of the first production airplane. If the economic life of the test article is reached prior to two lifetimes of durability testing, sufficient inspection in accordance with 5.3.2.3.a and b and data evaluation shall be completed prior to delivery of the first production airplane to estimate the extent of required production changes and retrofit. In the event the original schedule for the production decision and production delivery milestones become incompatible with the above schedule requirements, a study shall be conducted to assess the technical risk and cost impacts of changing these milestones. An important consideration in the durability test program is that it be completed at the earliest practical time. This is needed to minimize force modifications due to deficiencies found during testing. To this end the following needs to be accomplished:

a. Timely formulation of the test load spectra

b. Early delivery of the test article

c. Early establishment of managerial and contractual procedures for minimizing downtime in the event of a test failure.

Truncation, elimination, or substitution of load cycles in the test spectra to reduce test time and cost will be allowed. The contractor shall define by analysis and laboratory experiment the effect of any proposed truncation on the time to reach detrimental crack sizes to comply with the durability and damage tolerance requirements of MIL-A-8866 and MIL-A-83444 respectively. The results of these analyses and experiments shall be used to establish the final test spectra and, as necessary, to interpret the test results. The final test spectra shall require approval by the Air Force.

5.3.2.3 Inspections. Major inspection programs shall be conducted as an integral part of the full scale airframe durability test. The inspection programs shall require approval by the Air Force. These inspection programs shall include:

a. In-service design inspections developed in accordance with the damage tolerance requirements of MIL-A-83444 and the durability requirements of MIL-A-8866

b. Special inspections to monitor the status of critical areas and support the milestone schedule requirements of 5.3.2.2

c. Teardown inspection at the completion of the full scale durability test including any scheduled damage tolerance tests to support the interpretation and evaluation task of 5.3.8.

5.3.2.4 Test duration. The minimum durability test duration shall be as specified in MIL-A-8867. It may be advantageous to the Air Force to continue testing beyond the minimum requirement to determine life extension capabilities and validate design life capability for usage that is more severe than design usage. The decision to continue testing beyond the minimum duration shall be made based upon a joint rev[illegible] by the contractor and appropriate Air Force activities. The prospecti[illegible] ontractors shall provide, in their responses to the request for proposal, the estimated cost and schedule for two additional lifetimes of durability testing beyond the minimum requirement.

5.3.3 Damage tolerance tests. The contractor shall comply with the requirements for damage tolerance tests in MIL-A-8867 as specified in the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall require approval by the Air Force. The damage tolerance test program shall be of sufficient scope to verify Category I fracture critical parts in accordance with MIL-A-83444. The intent shall be to conduct damage tolerance tests on existing test hardware. This may include use of components and assemblies of the design development tests as well as the full scale static and durability test articles. When necessary, additional structural components and assemblies shall be fabricated and tested to verify compliance with the requirements of MIL-A-83444.

5.3.4 Flight and ground operations tests. The contractor shall comply with the detail requirements for flight and ground operations tests in MIL-A-8871 as specified in the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall require approval by the Air Force. An early Full Scale Development (FSD) or Research Development Test and Evaluation (RDT&E) airplane shall be used to perform the flight and ground operations tests. Load measurements shall be made by the strain gage or pressure survey method agreed to between the contractor and the Air Force. An additional airplane, sufficiently late in the production program to ensure obtaining the final configuration, shall be the backup airplane for these flight tests and shall be instrumented similar to the primary test aircraft. Special types of instrumentation (e.g., recording equipment, mechanical strain recorders, strain gages, etc.) to be used during the loads/environment spectra survey and the individual airplane tracking programs shall be placed on the structural flight test airplane as appropriate for evaluation and correlation. The flight and ground operations tests shall include a flight and ground loads survey and dynamic response tests.

5.3.4.1 Flight and ground loads survey. The flight and ground loads survey program shall consist of operating an instrumented and calibrated airplane within and to the extremes of its limit structural design envelope to measure the resulting loads and, if appropriate, to also measure pertinent temperature profiles on the airplane structure. The objectives of the loads survey shall be as follows:

a. Verification of the structural loads and temperature analysis used in the design of the airframe

b. Evaluation of loading conditions which produce the critical structural load and temperature distribution

c. Determination and definition of suspected new critical loading conditions which may be indicated by the investigations of structural flight conditions within the design limit envelope.

5.3.4.2 Dynamic response tests. The dynamic response tests shall consist of operating an instrumented and calibrated airplane to measure the structural loads and inputs while flying through atmospheric turbulence and during taxi, takeoff, towing, landing, refueling, store ejection, etc. The objectives shall be to obtain flight verification and evaluation of the elastic response characteristics of the structure to these dynamic load inputs for use in substantiating or correcting the loads analysis, fatigue analysis, and for interpreting the operational loads data.

5.3.5 Sonic durability tests The contractor shall comply with the detail requirements for sonic durability tests in MIL-A-8893 as specified in the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall require approval by the Air Force. Measurements shall be made of the acoustic environments on a full scale airplane to verify or modify the initial design acoustic loads/environment. The sonic durability test shall be conducted on a representative airplane (or its major components) to demonstrate structural adequacy for the design service life. Sonic durability tests normally are accomplished by ground testing of the complete airplane with the power plants operating at full power for a time sufficient to assure design service life. However, testing of major portions of the airplane in special nonreverberant ground test stands using the airplane propulsion system is the noise source, or in high intensity noise facilities, may be acceptable.

5.3.6 Flight vibration tests. The contractor shall comply with the detail requirements for flight vibration tests in MIL-A-8892 as specified in the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall require approval by the Air Force. These tests shall be conducted to verify the accuracy of the vibration analysis. In addition, the test results shall be used to demonstrate that vibration control measures are adequate to prevent cracking and to provide reliable performance of personnel and equipment throughout the design service life.

5.3.7 Flutter tests. The contractor shall comply with the detail requirements for flutter related tests in MIL-A-8870 as specified in the contract specifications. Prior to initiation of testing, the test plans, procedures, and schedules shall require approval by the Air Force. Flutter related tests shall consist of ground vibration tests, thermoelastic tests, limit load rigidity tests, control surface free play and rigidity tests, and flight flutter tests.

5.3.7.1 Ground vibration tests. The ground vibration tests shall consist of the experimental determination of the natural frequencies, mode shapes, and structural damping of the airframe or its components. The objective is to verify mass, stiffness, and damping characteristics which are used in the aeroelastic analyses (flutter analysis, dynamic analysis, math models, etc.).

5.3.7.2 Structural rigidity tests. The thermoelastic tests, limit load rigidity test, and control surface free play and rigidity tests shall consist of the experimental determination of the structural elastic and free play properties of the airframe and its components. The objective of these tests is to verify supporting data used in aeroelastic analyses and dynamic model design.

5.3.7.3 Flight flutter tests. Flight flutter tests shall be conducted to verify that the airframe is free from aeroelastic instabilities and has satisfactory damping throughout the operational flight envelope.

5.3.8 Interpretation and evaluation of test results. Each structural problem (failure, cracking, yielding, etc.) that occurs during the tests required by this standard shall be analyzed by the contractor to determine the cause, corrective actions, force implications, and estimated costs. The scope and interrelations of the various tasks within the interpretation and evaluation effort are illustrated in figure 4. The results of this evaluation shall define corrective actions required to demonstrate that the strength, rigidity, damage tolerance and durability design requirements are met. The cost, schedule, operational, and other impacts resulting from correction of deficiencies will be used to make major program decisions such as major redesign, program cancellation, awards or penalties, and production airplane buys. Structural modifications or changes derived from the results of the full scale test to meet the specified strength, rigidity, damage tolerance, and durability design requirements shall be substantiated by subsequent tests of components, assemblies, or full scale article as appropriate. (See figure 3.) The test duration for durability modifications shall be as specified in MIL-A-8867 and the contract specifications. The contractor shall propose these additional test requirements together with the associated rationale to the Air Force for approval.

5.4 Force management data package (Task V). Maintaining the strength, rigidity, damage tolerance, and durability is dependent on the capability of the appropriate Air Force commands to perform specific inspection, maintenance, and possibly modification or replacement tasks at specific intervals throughout the service life (i.e., at specified depot or base level maintenance times and special inspection periods). To properly perform these tasks, the Air Force must have detailed knowledge of the required actions. Additionally, experience has shown that the actual usage of military airplanes may differ significantly from the assumed design usage. It is necessary that the Air Force have the technical methods and actual usage data to assess the effect of these changes in usage on airplane damage tolerance and durability. Task IV describes the minimum required elements of a data package which the contractor shall provide so that the Air Force can accomplish the force management tasks as specified in 5.5. It should be noted that Task IV contains basic ASIP requirements to be performed by the contractor but, unlike Tasks I through III, is not for the purpose of providing compliance to the basic structural design requirements.

5.4.1 Final analyses. The contractor shall revise the design analyses as appropriate to account for significant differences between analysis and test that are revealed during the full scale tests and later during the loads/ environment spectra survey. These analyses updates shall be prepared as discussed below and shall require approval by the Air Force.

5.4.1.1 Initial update of analyses. The design analyses as specified in 5.2 shall be revised when the results of the design development and full scale tests as specified in 5.2.13 through 5.3.7 are available. These initial updates will be used to identify the causes of problems, corrective actions, and production and force modifications required by the interpretation and evaluation of test results task as specified in 5.3.8.

5.4.1.2 Final update of analyses. The initial update of the damage tolerance and durability analyses shall be revised to reflect the baseline operational spectra as specified in 5.4.3. These analysis updates shall form the basis for preparation of the updated force structural maintenance plan as specified in 5.4.3.2. The analyses shall identify the critical areas, damage growth rates, and damage limits required to establish the damage tolerance and durability inspection and modification requirements and economic life estimates required as part of the force structural maintenance plan.

5.4.1.3 Development of inspection and repair criteria. The appropriate analyses (stress, damage tolerance, durability, etc.) shall be used to develop a quantitative approach to inspection and repair criteria. Allowable damage limits and damage growth rates established by the analyses shall be used to develop inspection and repair times for structural components and assemblies. These analyses shall also be used to develop detail repair procedures for use at field or depot level. Special attention shall be placed on defining damage acceptance limits and damage growth rates for components utilizing bonded, honeycomb, or advanced composite types of construction. These inspection and repair criteria shall be incorporated into the force structural maintenance plan as specified in 5.4.3.

5.4.2 Strength summary. The contractor shall summarize the final analyses and other pertinent structures data into a format which will provide rapid visibility of the important structures characteristics, limitations and capabilities in terms of operational parameters. It is desirable that the summary be primarily in diagrammatic form showing the airplane structural limitations and capabilities as a function of the important operational parameters such as speed, acceleration, center of gravity location, and gross weight. The summary shall include brief descriptions of each major structural assembly, also preferably in diagrammatic form, indicating structural arrangements, materials, critical design conditions, damage tolerance and durability critical areas, and minimum margins of safety. Appropriate references to design drawings, detail analyses, test reports, and other back-up documentation shall be indicated. The strength summary shall require approval by the Air Force.

5.4.3 Force structural maintenance plan. The contractor shall prepare a force structural maintenance plan to identify the inspection and modification requirements and the estimated economic life of the airframe. Complete detailed information (when, where, how, and cost data as appropriate) shall be included. It is intended that the Air Force will use this plan to establish budgetary planning, force structure planning, and maintenance planning. This plan shall require approval by the Air Force.

5.4.3.1 Initial force structural maintenance plan. The initial plan shall be based on the design service life, design usage spectra, the results of the full scale test interpretation and evaluation task as specified in 5.3.7 and the upgraded critical parts list required as specified in 5.1.3.

5.4.3.2 Updated force structural maintenance plan. The force structural maintenance plan shall be updated to include the baseline operational spectra through use of the final analyses update as specified in 5.4.1.2. The first update of the plan shall be based on the analyses that utilized data obtained from the initial phase of the loads/environment spectra survey. Additional updates that may be required to reflect significant changes determined during continuation of the loads/environment spectra survey will be provided through separate negotiation between the Air Force and contractor.

5.4.4 Loads/environment spectra survey. The objective of the loads/environment spectra survey shall be to obtain time history records of those parameters necessary to define the actual stress spectra for the critical areas of the airframe. It is envisioned that 10-20 percent of the operational airplanes will be instrumented to measure such parameters as velocity, accelerations, altitude, fuel usage, temperature, strains, etc. The data will be obtained by the Air Force as part of the force management task as specified in 5.5 and shall be used by the contractor to construct the baseline operational spectrum as specified in 5.4.4.3. Data acquisition shall start with delivery of the first operational airplane. The contractor shall propose, in response to the request for proposal, the number of airplanes to be instrumented and the parameters to be monitored. For the purposes of the program definition, cost estimating, and scheduling, it shall be assumed that the duration of the survey will be 3 years or when the total recorded flight hours of unrestricted operational usage equals one design lifetime, whichever occurs first. The contractor shall also propose the method to be used to detect when a significant change in usage occurs to require an update in the baseline operational spectra. If the individual airplane tracking program as specified in 5.4.5 obtains sufficient data to develop the baseline operational spectra and detect significant usage changes, a separate survey program (or continuation thereof) as described herein may not be required. The scope of the program (e.g., the number of airplanes to be instrumented, and the number and type of parameters to be monitored) will be defined in the contract specifications.

5.4.4.1 Data acquisition provisions. The contractor shall select qualified functioning instrumentation and data recording systems in accordance with the requirements of this standard as specified in the contract specifications. The contractor shall select the specific instrumentation and data recording equipment to accomplish the survey task, obtain Air Force approval of the selections, and make the necessary instrumentation and data recording installations in the specified airplanes. If recording equipment and converter multiplexer equipment are selected, they shall meet the requirements of MIL-R-83165 and MIL-C-83166 respectively. Every effort should be made to use existing qualified instrumentation and recording equipment to reduce program costs and utilize proven operational capabilities. The contract shall specify whether the instrumentation and recording equipment (including spares) shall be Government Furnished Equipment (GFE) or Contractor Furnished Equipment (CFE).

5.4.4.2 Data processing provisions. The contractor shall coordinate with the Air Force the data processing provisions (including reformatting) to be used to ensure that the computer analysis methods will be compatible with the Air Force data analysis system. It is envisioned that contractor facilities and personnel, except for reformatting/transcribing and other data processing and analysis functions for which capabilities exist within the Air Force and are approved for use, will be used to process data collected during the 3-year period beginning with delivery of the first production airplane. Plans for transfer of data processing provisions from contractor to Air Force facilities including training of Air Force personnel shall be included.

5.4.4.3 Analysis of data and development of baseline operational spectra. The contractor shall use the flight data to assess the applicability of the design and durability test loads/environment spectra and to develop baseline operational spectra. The baseline operational spectra shall be used to update the durability and damage tolerance analyses as specified in 5.4.1.2 when a statistically adequate amount of data has been recorded. Subsequent revisions of the baseline operational spectra may be required but will require separate negotiations between the Air Force and contractor.

5.4.5 Individual airplane tracking program. The objective of the individual airplane tracking program shall be to predict the potential flaw growth in critical areas of each airframe that is keyed to damage growth limits of MIL-A-83444, inspection times, and economic repair times. Data acquisition shall start with delivery of the first operational airplane. The program shall include serialization of major components (e.g., wings, horizontal and vertical stabilizers, landing gears, etc.) so that component tracking can be implemented by the Air Force. The contractor shall propose for Air Force review and approval, an individual airplane tracking program for the specific airplane.

5.4.5.1 Tracking analysis method. The contractor shall develop an individual airplane tracking analysis method to establish and adjust inspection and repair intervals for each critical area of the airframe based on the individual airplane usage data. The damage tolerance and durability analyses and associated test data will be used to establish the analysis method. This analysis will provide the capability to predict crack growth rates, time to reach the crack size limits, and the crack length as a function of the total flight time and usage data. The contractor shall coordinate this effort with the Air Force to ensure that the computer analysis method will be compatible with the Air Force data analysis system. The individual airplane tracking analysis method shall require approval by the Air Force.

5.4.5.2 Data acquisition provisions. The contractor shall select qualified functioning instrumentation and data recording systems in accordance with the requirements of this standard as specified in the contract specifications. The recording system shall be as simple as possible and shall be the minimum required to monitor those parameters necessary to support the analysis methods as specified in 5.4.5.1. Counting accelerometers, electrical or mechanical strain recorders, electrical resistance gages, simplified manual data forms, etc. shall be considered. The contractor shall select the specific instrumentation and data recording equipment to accomplish the individual airplane usage tracking, obtain Air Force approval of the selections, and make the necessary instrumentation and data recording installations in the specified airplanes. The contract shall specify whether the instrumentation and recording equipment (including spares) shall be Government Furnished Equipment (GFE) or Contractor Furnished Equipment (CFE).

5.5 Force management (Task V). Task V describes those actions that must be conducted by the Air Force during force operations to ensure the damage tolerance and durability of each airplane. Task V will be primarily the responsibility of the Air Force and will be performed by the appropriate commands utilizing the data package supplied by the contractor in Task IV with the minimum amount of contractor assistance. Contractor responsibilities in Task V will be specified in the contract specifications.

5.5.1 Loads/environment spectra survey. The Air Force will be responsible for the overall planning and management of the loads/environment spectra survey and will:

a. Establish data collection procedures and transmission channels within the Air Force

b. Train squadron, base, and depot level personnel as necessary to ensure the acquisition of acceptable quality data

c. Maintain and repair the instrumentation and recording equipment

d. Ensure that the data are of acceptable quality and are obtained in a timely manner so that the contractor can analyze the results, develop the baseline spectrum (see 5.4.4.3), and update the analyses (see 5.4.1.2) and force structural maintenance plan (see 5.4.3.2).

The Air Force will also be responsible for ensuring that survey data are obtained for each type of usage that occurs within the force (training, reconnaissance, special tactics, etc.). Subsequent to completion of the initial data gathering effort, the Air Force will elect whether or not to continue to operate either all or a portion of the instrumentation and recording equipment aboard the survey airplanes to support additional updates of the baseline spectra and force structural maintenance plan.

5.5.2 Individual airplane tracking data. The Air Force will be responsible for the overall planning and management of the individual airplane tracking data gathering effort and will:

a. Establish data collection procedures and data transmission channels within the Air Force

b. Train squadron, base, and depot level personnel as necessary to ensure the acquisition of acceptable quality data

c. Maintain and repair the instrumentation and recording equipment

d. Ensure that the data are obtained and processed in a timely manner to provide adjusted maintenance times for each critical area of each airplane.

5.5.3 Individual airplane maintenance times. The Air Force will be responsible for deriving individual maintenance (inspection and repair) times for each critical area of each airplane by use of the tracking analysis methods as specified in 5.4.5.1 and the individual airplane tracking data as specified in 5.5.2. The objective is to determine adjusted times at which the force structural maintenance actions as specified in 5.4.3 have to be performed on individual airplanes and each critical area thereof. With the force structural maintenance plan and the individual aircraft maintenance time requirements available, the Air Force can schedule force structural maintenance actions on a selective basis that accounts for the effect of usage variations on structural maintenance intervals.

5.5.4 Structural maintenance records. AFLC and the using command will be responsible for maintaining structural maintenance records (inspection, repair, modification, and replacement) for individual airplanes. These records shall contain complete listings of structural maintenance actions that are performed with all pertinent data included (Time Compliance Technical Order (TCTO) action, component flight time, component and airplane serial number, etc.).

6. NOTES

6.1 Data requirements. The data requirements in support of this standard will be selected from the DOD Authorized Data List (TD-3) and will be reflected in a contractor data requirements list (DD Form 1423) attached to the request for proposal, invitation for bids, or the contract as appropriate.

6.2 Relationship to system engineering management. When appropriate, the conduct of the work efforts by the contractor in achieving airplane structural integrity will be included in the System Engineering Management Plan in accordance with MIL-STD-499A(USAF) for the airplane and will be compatible with the system safety plan in accordance with MIL-STD-882.

Custodian:
Air Force - 11

Review activities:
Air Force - 01, 10, 16

Preparing activity:
Air Force - 11

Project No. 15GP-F019

TABLE I. USAF Aircraft structural integrity program tasks.

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANALYSES AND DEVELOPMENT TESTS	FULL SCALE TESTING	FORCE MANAGEMENT DATA PACKAGE	FORCE MANAGEMENT
ASIP MASTER PLAN	MATERIALS AND JOINT ALLOWABLES	STATIC TESTS	FINAL ANALYSES	LOADS/ENVIRONMENT SPECTRA SURVEY
STRUCTURAL DESIGN CRITERIA	LOAD ANALYSIS	DURABILITY TESTS	STRENGTH SUMMARY	INDIVIDUAL AIRPLANE TRACKING DATA
DAMAGE TOLERANCE & DURABILITY CONTROL PLANS	DESIGN SERVICE LOADS SPECTRA	DAMAGE TOLERANCE TESTS	FORCE STRUCTURAL MAINTENANCE PLAN	INDIVIDUAL AIRPLANE MAINTENANCE TIMES
SELECTION OF MAT'LS, PROCESSES, & JOINING METHODS	DESIGN CHEMICAL/ THERMAL ENVIRONMENT SPECTRA	FLIGHT & GROUND OPERATIONS TESTS	LOADS/ENVIRONMENT SPECTRA SURVEY	STRUCTURAL MAINTENANCE RECORDS
DESIGN SERVICE LIFE AND DESIGN USAGE	STRESS ANALYSIS	SONIC TESTS	INDIVIDUAL AIRPLANE TRACKING PROGRAM	
	DAMAGE TOLERANCE ANALYSIS	FLIGHT VIBRATION TESTS		
	DURABILITY ANALYSIS	FLUTTER TESTS		
	SONIC ANALYSIS	INTERPRETATION & EVALUATION OF TEST RESULTS		
	VIBRATION ANALYSIS			
	FLUTTER ANALYSIS			
	NUCLEAR WEAPONS EFFECTS ANALYSIS			
	NON-NUCLEAR WEAPONS EFFECTS ANALYSIS			
	DESIGN DEVELOPMENT TESTS			

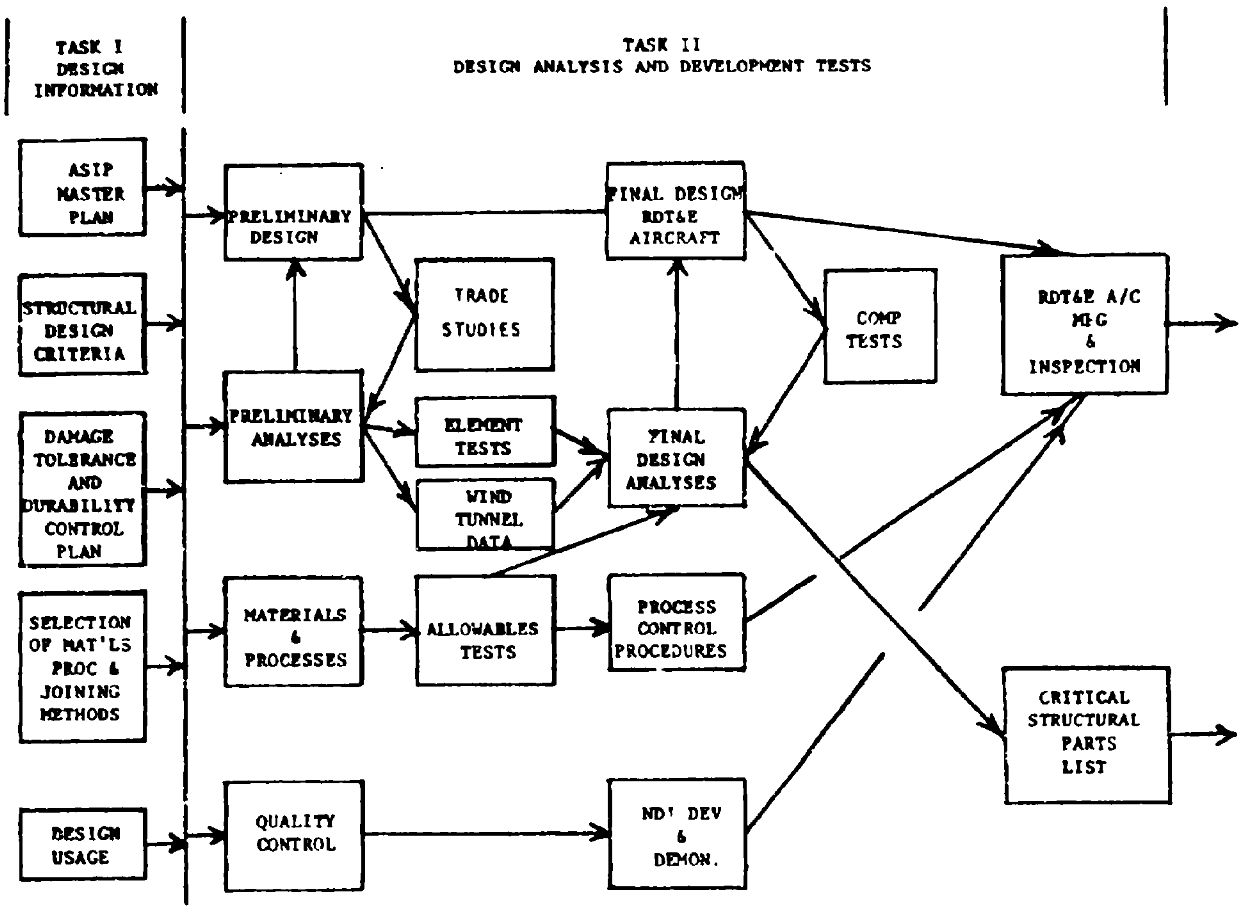


FIGURE 1. Aircraft structural integrity program.

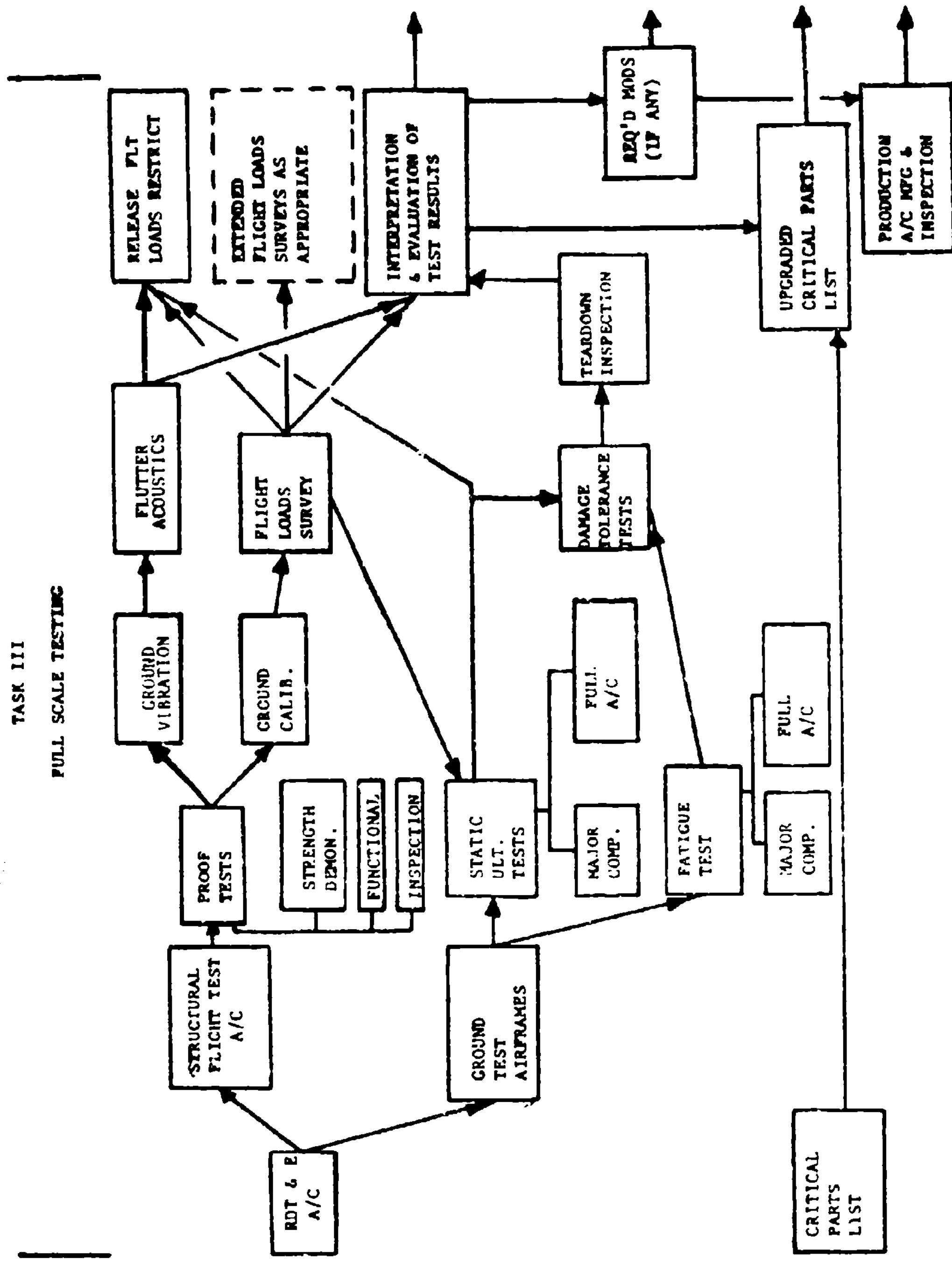


FIGURE 2. Aircraft structural integrity program.

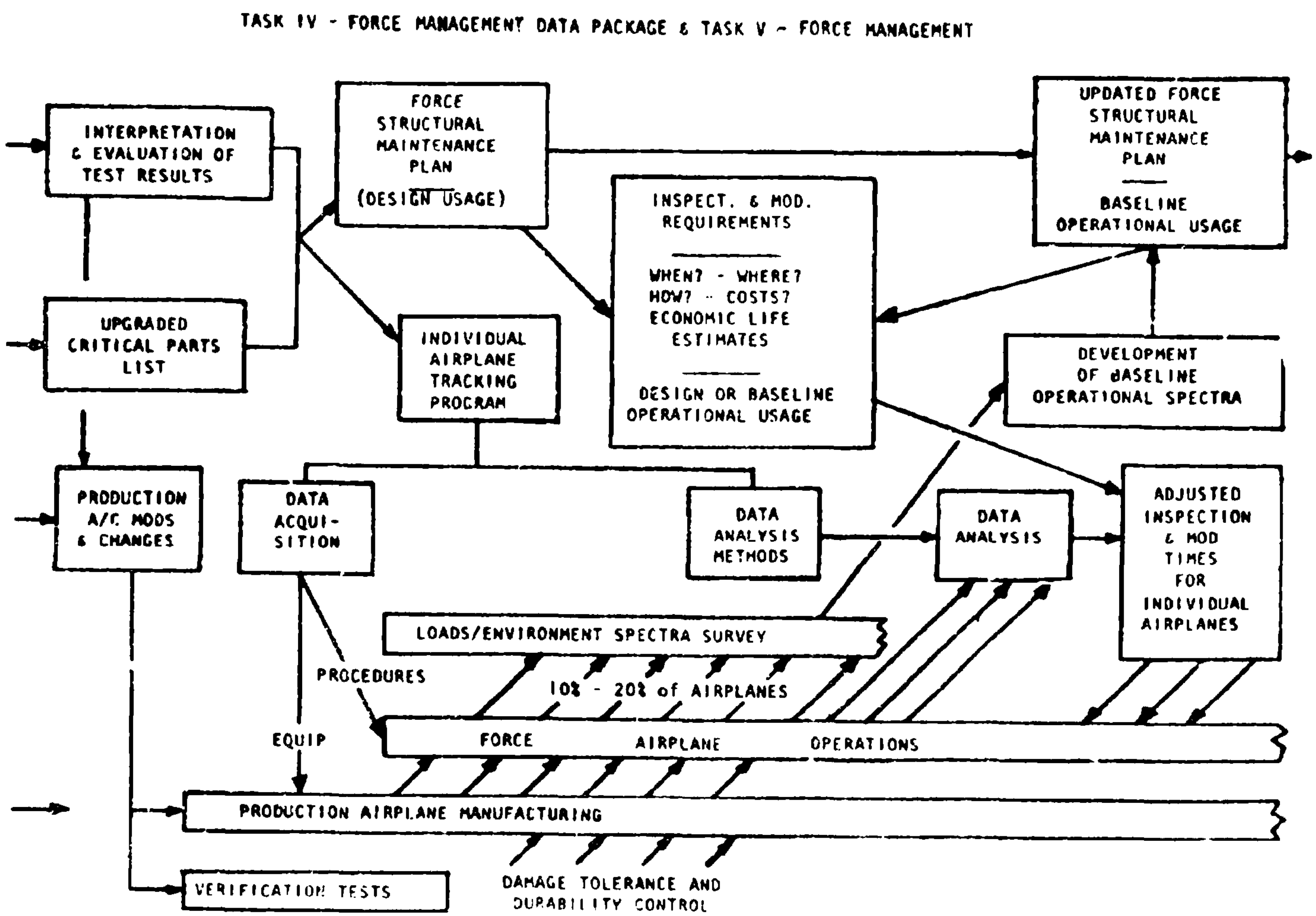


FIGURE 3. Aircraft structural integrity program.

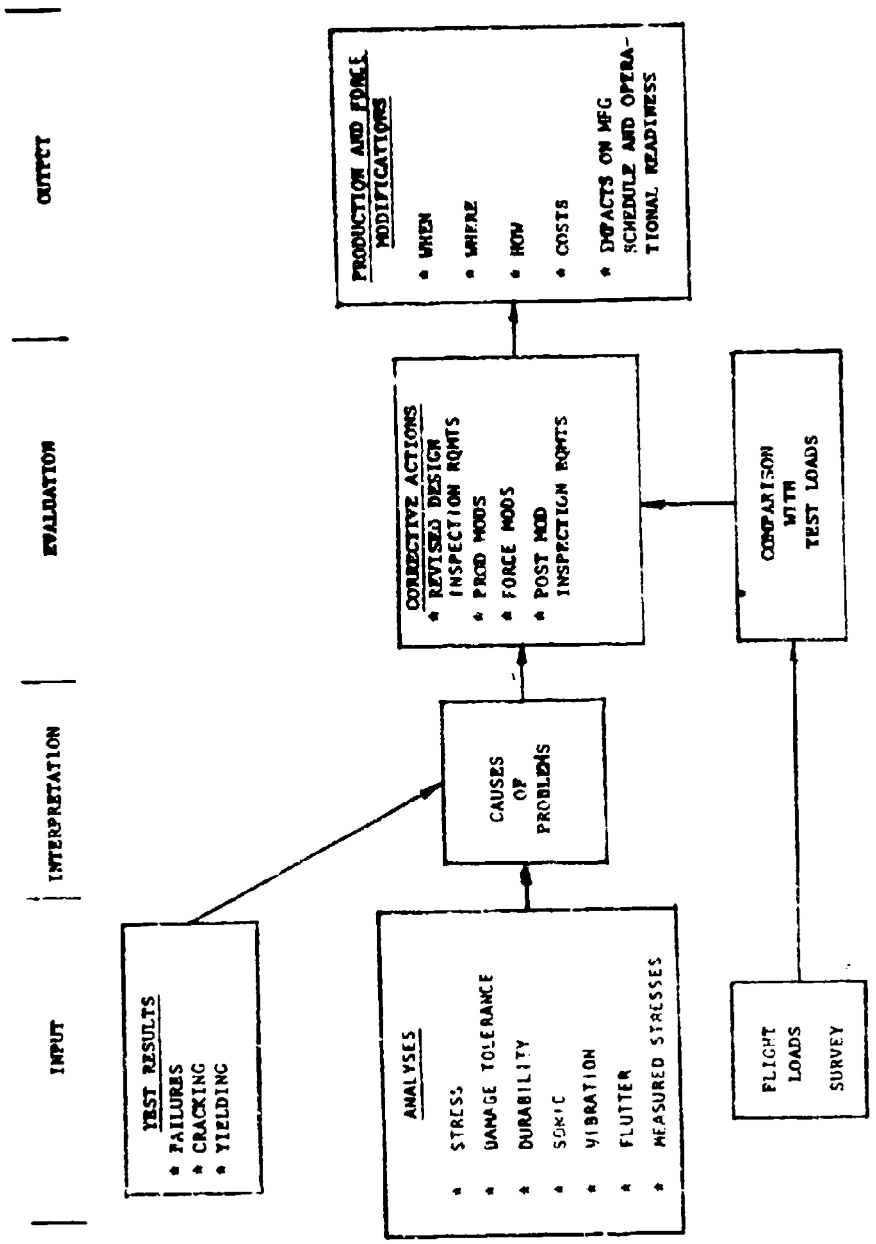


FIGURE 4. Interpretation and evaluation of test results (based on design service life and design usage).

STANDARDIZATION DOCUMENT IMPROVEMENT PROPOSAL

OMB Approval
No. 22-R255

INSTRUCTIONS: The purpose of this form is to solicit beneficial comments which will help achieve procurement of suitable products at reasonable cost and minimum delay, or will otherwise enhance use of the document. DoD contractors, government activities, or manufacturers/vendors who are prospective suppliers of the product are invited to submit comments to the government. Fold on lines on reverse side, staple in corner, and send to preparing activity. Comments submitted on this form do not constitute or imply authorization to waive any portion of the referenced document(s) or to amend contractual requirements. Attach any pertinent data which may be of use in improving this document. If there are additional papers, attach to form and place both in an envelope addressed to preparing activity.

DOCUMENT IDENTIFIER AND TITLE

NAME OF ORGANIZATION AND ADDRESS

CONTRACT NUMBER

MATERIAL PROCURED UNDER A

☐ DIRECT GOVERNMENT CONTRACT ☐ SUBCONTRACT

1. HAS ANY PART OF THE DOCUMENT CREATED PROBLEMS OR REQUIRED INTERPRETATION IN PROCUREMENT USE?

A. GIVE PARAGRAPH NUMBER AND WORDING.

B. RECOMMENDATIONS FOR CORRECTING THE DEFICIENCIES

2. COMMENTS ON ANY DOCUMENT REQUIREMENT CONSIDERED TOO RIGID

3. IS THE DOCUMENT RESTRICTIVE?

☐ YES ☐ NO *(If "Yes", in what way?)*

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MILITARY SPECIFICATION

AIRPLANE DAMAGE TOLERANCE REQUIREMENTS

This specification is approved for use by all Departments and Agencies of the Department of Defense.

1. SCOPE

1.1 This specification contains the damage tolerance design requirements applicable to airplane safety of flight structure. The objective is to protect the safety of flight structure from potentially deleterious effects of material, manufacturing and processing defects through proper material selection and control, control of stress levels, use of fracture resistant design concepts, manufacturing and process controls and the use of careful inspection procedures.

2. APPLICABLE DOCUMENTS

2.1 The following documents, of the issue in effect on date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein:

SPECIFICATIONS

Military

MIL-A-8866	Airplane Strength and Rigidity, Ground Tests
MIL-A-8867	Airplane Strength and Rigidity, Reliability Requirements, Repeated Loads and Fatigue

STANDARDS

Military

MIL-STD-1530	Aircraft Structural Integrity Program Airplane Requirements

(Copies of documents required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

FSC 1510

3. REQUIREMENTS

3.1 General requirements. Detailed damage tolerance requirements are specified in 3.2 in various categories as a function of design concept and degree of inspectability. The design concepts and various degrees of inspectability are defined in 6.2. The contractor shall demonstrate that all safety of flight structures comply with the detailed requirements in a minimum of one of these categories (one design concept and one inspectability level). Design concepts utilizing multiple load paths and crack arrest features may be qualified under the appropriate inspectability level(s) as either slow crack growth or fail-safe structure. Single-load path structure without crack arrest features must be qualified at the appropriate inspectability level(s) as slow crack growth structure. The contractor shall perform all of the analytical and experimental work necessary to demonstrate compliance with the damage tolerance analyses and tests as specified herein, MIL-STD-1530, MIL-A-8867 and the procurement contract. This effort involves residual strength and crack growth analyses and tests. The analyses shall assume the presence of flaws placed in the most unfavorable location and orientation with respect to the applied stresses and material properties. The crack growth analyses shall predict the growth behavior of these flaws in the chemical, thermal, and sustained and cyclic stress environments to which that portion of the component shall be subjected in service. The design flight by flight stress spectra and chemical and thermal environment spectra shall be developed by the contractor and approved by the procuring activity. Spectra interaction effects, such as variable loading and environment, shall be accounted for.

3.1.1 Initial flaw assumptions. Initial flaws shall be assumed to exist as a result of material and structure manufacturing and processing operations. Small imperfections equivalent to an .005 inch radius corner flaw resulting from these operations shall be assumed to exist in each hole of each element in the structure, and provide the basis for the requirements in 3.1.1.1c, 3.1.1.2, 3.1.1.3.1 and 3.1.1.3.2. If the contractor has developed initial quality data on fastener holes, (e.g., by fractographic studies, which provides a sound basis for determining equivalent initial flaw sizes), these data may be submitted to the procuring activity for review and serve as a basis for negotiating a size different than the specified .005 inch radius corner flaw. In addition, it shall be assumed that initial flaws of the sized specified in 3.1.1.1a, and b, can exist in any separate element of the structure. Each element of the structure shall be surveyed to determine the most critical location for the assumed initial flaws considering such features as edges, fillets, holes and other potentially high stressed areas. Only one initial flaw in the most critical hole and one initial flaw at a location other than a hole need be assumed to exist in any structural element. Interaction between the assumed initial flaws need not be considered. For multiple and adjacent elements, the initial flaws need not be situated at the same location, (e.g., stacked plates in wing structure), except for structural elements where fabrication and assembly operations are conducted such that flaws in two or more elements can exist at the same location. The most common example of such an operation is the assembly drilling of attachment holes. Except as noted in 3.1.1.2 and 3.1.1.3 more than one source of common initial cracks need not be assumed along the crack growth path. Initial flaw sizes are specified in terms of specific flaw shapes, such as through the thickness or corner flaws at holes and semi-elliptical surface flaws or through the thickness flaws at locations other than holes.

3.1.1.1 Initial flaw size. Specified initial flaw sizes presume the inspection of 100 percent of all fracture critical regions of all structural components as required by the fracture control provisions of MIL-STD-1530. This inspection shall include as a minimum a close visual inspection of all holes and cutouts and conventional ultrasonic, penetrant or magnetic particle inspection of the remainder of the fracture critical region. Where the use of automatic hole preparation and fastener installation equipment preclude close visual and dimensional inspection of 100 percent of the holes in the fracture critical regions of the structure, a plan to qualify and monitor hole preparation and fastener installation shall be prepared, approved by the procuring activity and implemented by the contractor. Where special nondestructive inspection procedures have demonstrated a detection capability better than indicated by the flaw sizes specified in (a) below, and the resulting smaller assumed flaw sizes are used in the design of the structure, these special inspection procedures shall be used in the aircraft manufacturing quality control.

a. Slow crack growth structure.

At holes and cutouts the assumed initial flaw shall be a .05 inch through the thickness flaw at one side of the hole when the material thickness is equal to or less than .05 inch. For material thicknesses greater than .05 inch, the assumed initial flaw shall be a .05 inch radius corner flaw at one side of the hole.

At locations other than holes, the assumed initial flaw shall be a through the thickness flaw .25 inch in length when the material thickness is equal to or less than .125 inch. For material thicknesses greater than .125 inch, the assumed initial flaw shall be a semicircular surface flaw with a length (2c) equal to .25 inch and a depth (a) equal to .125 inch. Other possible surface flaw shapes with the same initial stress intensity factor (K) shall be considered as appropriate. For example, corner flaws at edges of structural elements and longer and shallower surface flaws in plates which are subjected to high bending stresses.

Smaller initial flaw sizes than those specified above may be assumed subsequent to a demonstration, described in 4.2, that all flaws larger than these assumed sizes have at least a 90 percent probability of detection with a 95 percent confidence level. Smaller initial flaw sizes may also be assumed if proof test inspection is used. In this case, the minimum assumed initial flaw size shall be the calculated critical size at the proof test stress level and temperature using procuring activity approved upper bound of the material fracture toughness data.

b. Fail safe structure.

At holes and cutouts the assumed initial flaw shall be a .02 inch through the thickness flaw at one side of the hole when the material thickness is equal to or less than .02 inch. For material thicknesses greater than .02 inch the assumed initial flaw shall be a .02 inch radius corner flaw at one side of the hole.

At locations other than holes, the assumed initial flaw shall be a through the thickness flaw .10 inch in length when the material thickness is equal to or less than .05 inch. For material thicknesses greater than .05 inch, the assumed initial flaw shall be a semicircular surface flaw with a length (2c) equal to .10 inch and a depth (a) equal to .05 inch. Other possible surface flaw shapes with the same initial stress intensity factor (K) shall be considered as appropriate.

c. The fastener policy.

The beneficial effects of interference fasteners, cold expanded holes, joint clamp-up and other specific joint design and assembly procedures may be used in achieving compliance to the flaw growth requirements of this specification. These beneficial effects shall be demonstrated by laboratory tests of joints representative of the joints in the aircraft. The test specimens shall contain pre-cracked fastener holes. The limits of the beneficial effects to be used in design shall be approved by the procuring activity, but in no case shall the assumed initial flaw be smaller than an .005 inch radius corner flaw at one side of an as manufactured, non-expanded hole containing a net fit fastener in a non-clamped-up joint.

3.1.1.2 Continuing damage. Cyclic growth behavior of assumed initial flaws may be influenced by the particular geometry and arrangement of elements of the structure being qualified. The following assumptions of continuing crack growth shall be considered for those cases where the primary crack terminates due to structural discontinuities or element failure.

a. When the primary damage and growth originates in a fastener hole and terminates prior to member or element failure, continuing damage shall be an .005 inch radius corner flaw emanating from the diametrically opposite side of the fastener hole at which the initial flaw was assumed to exist.

b. When the primary damage terminates due to a member or element failure, the continuing damage shall be an .005 inch radius corner flaw in the most critical location of the remaining element or remaining structure or a surface flaw having 2c = .02 inch and a = .01 inch, where, a, is measured in the direction of crack growth, plus the amount of growth (Δa) which occurs prior to element failure.

c. When the crack growth from the assumed initial flaw enters into and terminates at a fastener hole, continuing damage shall be an .005 inch radius corner flaw +Δa emanating from the diametrically opposite side of the fastener hole at which the primary damage terminated.

3.1.1.3 Remaining structure damage

3.1.1.3.1 Fail safe multi-load path. The damage assumed to exist in the adjacent load path at the location of primary failure in fail safe multiple load path structure at the time of and subsequent to the failure of a primary load path shall be as follows:

a. Multiple load path dependent structure. The same as specified in 3.1.1.1b plus the amount of growth (Δa) which occurs prior to load path failure.

b. Multiple load path independent structure. The same as 3.1.1.2b plus the amount of growth (Δa) which occurs prior to load path failure.

3.1.1.3.2 Fail safe crack arrest structure. For structure classified as fail safe-crack arrest, the primary damage assumed to exist in the structure following arrest of a rapidly propagating crack shall depend upon the particular geometry. In conventional skin stringer (or frame) construction this shall be assumed as two panels (bays) of cracked skin plus the broken central stringer (or frames). Where tear straps are provided between stringers (or frames) this damage shall be assumed as cracked skin between tear straps plus the broken central stringer (or frame). Other configurations shall assume equivalent damage as mutually agreed upon by the contractor and the procuring activity. The damage assumed to exist in the structure adjacent to the primary damage shall be as specified in 3.1.1.2b or c.

3.1.2 In-Service inspection flaw assumptions. The smallest damage which can be presumed to exist in the structure after completion of a depot or base level inspection shall be as follows:

a. If the component is to be removed from the aircraft and completely inspected with NDI procedures the same as those performed during fabrication, the minimum assumed damage size shall be as specified in 3.1.1.1.

b. Where NDI techniques such as penetrant, magnetic particle or ultrasonics are applied without component or fastener removal, the minimum assumed flaw size at holes and cutouts shall be a through the thickness crack emanating from one side of the hole having a 0.25 inch uncovered length when the material thickness is equal to or less than 0.25 inch. For material thicknesses greater than 0.25 inch, the assumed initial flaw shall be a quarter-circular corner crack emanating from one side of the hole having a 0.25 inch uncovered length. The minimum assumed flaw size at locations other than holes shall be a through the thickness crack of length 0.50 inch when the material thickness is equal to or less than 0.25 inch. For material thicknesses greater than 0.25 inch, the assumed initial flaw shall be a semi-circular surface flaw with length (2c) equal to 0.50 inch and depth (a) equal to 0.25 inch. Other possible surface flaw shapes with the same initial stress intensity factor (K) shall be considered as appropriate such as corner flaws at edges of structural members and longer and shallower surface flaws in plates which are subjected to high bending stresses. While X-ray inspection may be used to supplement one or more of the other NDI techniques, it, by itself, shall not be considered capable of reliably detecting tight subcritical cracks.

c. Where accessibility allows close visual inspection (using visual aid as necessary) an opening through the thickness crack having at least 2 inches of uncovered length shall be the minimum assumed damage size.

d. Where accessibility, paint, sealant, or other factors preclude close visual inspection or the use of NDI techniques such as described in b above, slow crack growth structure shall be considered to be non-inspectable, and fail-safe structure shall be considered to be inspectable only for major damage such as a load path failure, or arrested unstable crack growth.

3.1.3 Residual strength requirements. The minimum required residual strength is specified in terms of the minimum internal member load, P_{XX}, which the aircraft must be able to sustain with damage present and without endangering safety of flight or degrading performance of the aircraft for the specified minimum period of unrepaired service usage. This includes loss of strength, loss of stiffness, excessive permanent deformation, loss of control, and reduction of the flutter speed below V_L. The magnitude of P_{XX} depends on the overall degree of inspectability of the structure and is intended to represent the maximum load the aircraft might encounter during a specified inspection interval or during a design lifetime for non-inspectable structure. The XX subscript is defined as a function of the specific degree of inspectability in table I.

The aircraft loading spectrum shall be derived from a mission analysis where the mission mix and the loads in each mission segment represent average aircraft usage. The basic load factor exceedance data is specified in MIL-A-8866. To account for the fact that any individual aircraft may encounter loads considerably in excess of the average during their life, the required residual strength (i.e. the P_{XX} load) must be larger than the average load expected during a given interval between inspections. This is accomplished by magnifying the inspection interval. For example, the P_{XX} load for ground evident damage is the maximum average load that can be expected once in 100 flights. Table 1 defines the P_{XX} loads for the various degrees of inspectability.

TABLE 1

P_{XX}*	Degree of Inspectability	Typical Inspection Interval	Magnification Factor, M
P_{FE}	In-Flight Evident	One Flight	100
P_{GE}	Ground Evident	One Flight	100
P_{WV}	Walk-Around Visual	Ten Flights	100
P_{SV}	Special Visual	One Year	50
P_{DM}	Depot or Base Level	1/4 Lifetime	20
P_{LT}	Non-Inspectable	One Lifetime	20

*P_{XX} = Maximum average internal member load that will occur once in M times the inspection interval. Where P_{DM} or P_{LT} is determined to be less than the design limit load, the design limit load shall be the required residual strength load level. P_{XX} need not be greater than 1.2 times the maximum load in one lifetime, if greater than design limit load.

For fail safe structure there is a requirement to sustain a minimum load, P_{YY}, at the instant of load path failure (or crack arrest) as well as being able to sustain the load, P_{XX}, subsequent to load path failure (or crack arrest) at any time during the specified inspection interval. The single load path failure (or crack arrest) load, P_{YY}, shall include a dynamic factor (D.F.). In lieu of test or analytical data to the contrary a dynamic factor of 1.15 shall be applied to the redistributed incremental load. P_{YY} should be equal to the internal member load at design limit load or D.F. times P_{XX}, whichever is greater.

3.2 Specific requirements. Specific damage tolerance requirements for Slow crack growth structure, Fail-Safe multiple load path structure, and Fail-Safe crack arrest structure as specified in 3.2.1, 3.2.2 and 3.2.3, respectively.

3.2.1 Slow crack growth structure. Of the degrees of inspectability in accordance with 6.2.1, only depot or base level inspectable and in-service non-inspectable are applicable to slow crack growth structures. The frequency of inspection for both shall be as stated below unless otherwise specified in the appropriate contractual documents.

Depot or Base Level inspectable - Once every one quarter of the design lifetime.

In-Service non-inspectable - Once at the end of one design lifetime.

3.2.1.1 Depot or base level inspectable. The damage which can be presumed to exist in the structure after completion of a depot or base level inspection shall be that specified for slow crack growth structure in 3.1.2. These damage sizes shall not grow to critical size and cause failure of the structure due to the application of P_{DM} in two (2) times the inspection interval as specified in 3.2.1.

3.2.1.2 In-Service non-inspectable. The initial damage size as specified in 3.1.1.1 shall not grow to critical size and cause failure of the structure due to the application of P_{LT} in two (2) design service lifetimes.

3.2.2 Fail-Safe multiple load path structure. The degrees of inspectability as specified in 6.2.1, which can be applicable to fail-safe multiple load path structure, are In-Flight evident inspectable, Ground evident inspectable, Walkaround inspectable, Special visual inspectable, and Depot or Base level inspectable.

3.2.2.1 Inspection intervals. The frequency of inspection for each of the inspectability levels shall be as stated below unless otherwise specified in the appropriate contractual documents.

In-Flight evident inspectable - Once per flight.

Ground evident inspectable - Once per flight.

Walkaround inspectable - Once every ten (10) flights.

Special visual inspectable - As proposed by the contractor and approved by the procuring activity, but not more frequently than once per year.

Depot or Base level inspectable - Once every one quarter of the design lifetime.

3.2.2.2 Residual strength requirements and damage growth limits. There are two sets of residual strength requirements and damage growth limits for fail-safe multiple load path structure. The first set applies to the required residual strength and damage growth limits for intact structure, (i.e., the structure prior to a load path failure), and the second set applies to the remaining structure subsequent to a load path failure. These are described in 3.2.2.2.1 and 3.2.2.2.2, respectively, and are summarized in table II.

3.2.2.2.1 Intact structure. The requirements for the intact structure are a function of the depot or base level inspectability of the intact structure for damage sizes which are less than a load path failure, (i.e., subcritical cracks and small element failures). If the structure is depot or base level inspectable the smallest damage sizes which can be presumed to exist in the structure after completion of a depot or base level inspection shall be those as specified in 3.1.2. These damage sizes shall not grow to a size such as to cause load path failure due to the application of P_{DM} in one depot or base level inspection interval. If the structure is not depot or base level inspectable for subcritical flaws or small element failures which are less than a load path failure (either by virtue of small critical flaw sizes or inspection problems) the initial material and manufacturing damage as specified in 3.1.1.1.b shall be assumed and it shall not grow to the size required to cause load path failure due to the application of P_{LT} in one design lifetime.

3.2.2.2.2 Remaining structure subsequent to a load path failure. For each of the five levels of inspectability specified in 3.2.2 the remaining structure at the time of a load path failure shall be able to sustain the P_{YY} load as described in 3.1.3 without loss of the aircraft. In addition, subsequent to load path failure, the failed load path plus the minimum assumed damage in the remaining adjacent structure as specified in 3.1.1.3.1 shall not grow to a size such as to cause loss of the aircraft due to the application of the P_{XX} load in the specified minimum period of unrepaired service usage. The P_{XX} loads and minimum periods of unrepaired service usage for each of the five inspectability levels shall be as follows:

Inspectability	P_{XX} per 3.1.3	Minimum Period of Unrepaired Service Usage
In-Flight Evident	P_{FE}	Return to base
Ground Evident	P_{GE}	One Flight
Walkaround	P_{WV}	5 X Inspection Interval*
Special Visual	P_{SV}	2 X Inspection Interval*
Depot or Base Level	P_{DM}	2 X Inspection Interval*

*See 3.2.2.1

TABLE II

FAIL-SAFE MULTIPLE LOAD PATH STRUCTURE

INSPECTABILITY	Residual strength req. and damage growth limits for intact structure	Residual strength req. and damage growth limits for remaining structure subsequent to load path failure
IN-FLIGHT EVIDENT	○ If Structure is Depot or Base Level Inspectable for less than Failed Load Path (e.g. subcritical flaws):	○ Must sustain P_{YY} at time of load path failure ○ ℓ shall not cause aircraft failure @ P_{FE} during return to base
GROUND EVIDENT	$(a)_{DM}$ shall not grow critical @ P_{DM} in one Depot or Base Level inspection interval	○ Must sustain P_{YY} at time of failure. ○ ℓ shall not cause aircraft failure @ P_{GE} in one flight
WALKAROUND VISUAL	or ○ If Structure is not Depot or Base Level inspectable for Less than Failed Load Path:	○ Must sustain P_{YY} at time of failure ○ ℓ shall not cause aircraft failure @ P_{WV} in 5 times the inspection interval
SPECIAL VISUAL	a_i shall not grow to critical @ P_{LT} in one lifetime	○ Must sustain P_{YY} at time of failure ○ ℓ shall not cause aircraft failure @ P_{SV} in 2 times the inspection interval
DEPOT OR BASE LEVEL		○ Must sustain P_{YY} at time of failure ○ ℓ shall not cause aircraft failure @ P_{DM} in 2 times the inspection interval

$(a)_{DM}$ = Assumed Depot or Base Level Damage Sizes specified in 3.1.2

a_i = Assumed Initial flaw sizes specified in 3.1.1.1.b

ℓ = Failed Load Path plus assumed damage in remaining structure as specified in 3.1.1.3.1

P_{YY} = D.F. X P_{LIMIT} or D.F. X P_{XX} whichever is greater

3.2.3 Fail-Safe crack arrest structure. The degrees of inspectability as specified in 6.2.1 which can be applicable to fail-safe crack arrest structure are the same as those for the fail-safe multiple load path structure specified in 3.2.2.

3.2.3.1 Inspection intervals. The frequency of inspection for each of the inspectability levels shall be the same as those specified for fail-safe multiple load path structure in 3.2.2.1.

3.2.3.2 Residual strength requirements and damage growth limits. There are two sets of residual strength requirements and damage tolerance limits for fail-safe crack arrest structure. The first set applies to the intact structure (the structure prior to unstable crack growth and arrest equivalent to that as specified in 3.1.1.3.2) and the second set applies to the remaining structure subsequent to encountering unstable crack growth and arrest. These are described in 3.2.3.2.1 and 3.2.3.2.2, respectively and are summarized in table III.

3.2.3.2.1 Intact structure. The requirements for the intact structure are a function of the depot or base level inspectability of the intact structure for damage sizes which are less than the damage caused by unstable crack growth and arrest as specified in 3.1.1.3.2. If the structure is depot or base level inspectable the smallest damage sizes which can be presumed to exist in the structure after completion of a depot or base level inspection shall be those as specified in 3.1.2. These sizes shall not grow to a size such as to cause unstable crack growth due to the application of P_{DM} in one depot or base level inspection interval. If the structure is not depot or base level inspectable for subcritical flaws, the initial material and manufacturing damage as specified in 3.1.1.1.b shall be assumed and it shall not grow to critical size at P_{LT} in one design lifetime.

3.2.3.2.2 Remaining structure subsequent to crack arrest. For each of the five levels of inspectability applicable to this type of structure the remaining structure at the time of the unstable crack growth shall be able to sustain the P_{YY} load as specified in 3.1.3 without loss of the aircraft. In addition, subsequent to the unstable growth and arrest, damage as specified in 3.1.1.3.2 shall not grow to a size such as to cause loss of the aircraft due to the application of the P_{XX} load in the specified minimum periods of unrepaired usage. The P_{XX} loads and minimum periods of unrepaired service usage for each of the five inspectability levels shall be the same as those specified for fail-safe multiple load path structure in 3.2.2.2.2.

TABLE III

FAIL-SAFE CRACK ARREST STRUCTURE

INSPECTABILITY	Residual strength req. and damage growth limits for intact structure	Residual strength req. and damage growth limits for remaining structure subsequent to unstable growth and arrest
IN-FLIGHT EVIDENT	○ If Structure is Depot or Base Level Inspectable for less than arrested Damage (e.g. subcritical flaws):	○ Must sustain P_{YY} at time of unstable cracking ○ ℓ' shall not cause A/C failure @ P_{FE} during return to base
GROUND EVIDENT	$(a)_{DM}$ shall not grow to critical @ P_{DM} in one or Base level inspection interval	○ Must sustain P_{YY} at times of unstable cracking ○ ℓ' shall not cause A/C failure @ P_{GE} in one flight
WALKAROUND VISUAL	or	○ Must sustain P_{YY} at time of unstable cracking ● ℓ' shall not cause A/C failure @ P_{WV} in 5 times the inspection interval
SPECIAL VISUAL	○ If Structure is not Depot or Base Level Inspectable for less than Arrested Dar : a_i shall not grow to critical @ P_{LT} in one lifetime	○ Must sustain P_{YY} at time of unstable cracking ● ℓ' shall not cause A/C failure @ P_{SV} in 2 times inspection interval
DEPOT OR BASE LEVEL		○ Must sustain P_{YY} at time of unstable cracking ○ ℓ' shall not cause A/C Failure @ P_{DM} in 2 times inspection interval

$(a)_{DM}$ = Assumed Depot or Base Level Damage Sizes specified in 3.1.2

a_i = Assumed initial flaw sizes specified in 3.1.1.1.b

ℓ' = Damage as specified in 3.1.1.3.2

P_{YY} = D.F. X P_{LIMIT} or D.F. P_{XX} whichever is greater

4. QUALITY ASSURANCE PROVISIONS

4.1 Design data. Design data shall be generated as required to support the analysis effort.

4.2 NDT demonstration program. Where designs are based on initial flaw size assumptions less than those as specified in 3.1.1.1a, a non-destructive testing demonstration program shall be performed by the contractor and approved by the procuring activity to verify that all flaws equal to or greater than the design flaw size will be detected to the specified reliability and confidence levels. The demonstration shall be conducted on each selected inspection procedure using production conditions, equipment and personnel. The defective hardware used in the demonstration shall contain cracks which simulate the case of tight fabrication flaws. Subsequent to successful completion of the demonstration program, specifications on these inspection techniques shall become the manufacturing inspection requirements and may not be changed without a requalifying program subject to procuring activity approval.

5. PREPARATION FOR DELIVERY

NOT APPLICABLE

6. NOTES

6.1 Intended use. This specification is intended for use in the design of all new military airplanes for the procuring activity. It is not intended to be directly applicable to advanced composite structures nor landing gear components. It is also not intended to dictate structural design concepts, however, other requirements that may be imposed on specific aircraft systems (battle or foreign object damage requirements) may limit the choices.

6.2 Definitions

6.2.1 Degree of inspectability. The degree of inspectability of safety of flight structure shall be established in accordance with the following definitions.

6.2.1.1 In-Flight evident inspectable. Structure is in-flight evident inspectable if the nature and extent of damage occurring in flight will result directly in characteristics which make the flight crew immediately and unmistakably aware that significant damage has occurred and that the mission should not be continued.

6.2.1.2 Ground evident inspectable. Structure is ground evident inspectable if the nature and extent of damage will be readily and unmistakably obvious to ground personnel without specifically inspecting the structure for damage.

6.2.1.3 Walkaround inspectable. Structure is walkaround inspectable if the nature and extent of damage is unlikely to be overlooked by personnel conducting a visual inspection of the structure. This inspection normally shall be a visual look at the exterior of the structure from ground level without removal of access panels or doors without special inspection aids.

6.2.1.4 Special visual inspectable. Structure is special visual inspectable if the nature and extent of damage is unlikely to be overlooked by personnel conducting a detailed visual inspection of the aircraft for the purpose of finding damaged structure. The procedures may include removal of access panels and doors, and may permit simple visual aids such as mirrors and magnifying glasses. Removal of paint, sealant, etc, and use of NDI techniques such as penetrant, X-ray, etc. are not part of a special visual inspection.

6.2.1.5 Depot or base level inspectable. Structure is depot or base level inspectable if the nature and extent of damage will be detected utilizing one or more selected nondestructive inspection procedures. The inspection procedures may include NDI techniques such as penetrant, X-ray, ultrasonic, etc. Accessibility considerations may include removal of those components designed for removal.

6.2.1.6 In-Service non-inspectable structure. Structure is in-service non-inspectable if either damage size or accessibility preclude detection during one or more of the above inspections.

6.2.2 Frequency of inspection. Frequency of inspection is the number of times that a particular type of inspection is to be conducted during the service life of the aircraft.

6.2.3 Minimum period of unrepaired service usage. Minimum period of unrepaired service usage is that period of time during which the appropriate level of damage (assumed initial or in-service) is presumed to remain unrepaired and allowed to grow within the structure.

6.2.4 Minimum assumed initial damage size. The minimum assumed initial damage size is the smallest crack-like defect which shall be used as a starting point for analyzing residual strength and crack growth characteristics of the structure.

6.2.5 Safety of flight structure. That structure whose failure could cause direct loss of the aircraft, or whose failure if it remained undetected could result in loss of the aircraft.

6.2.6 Fracture critical structure. Safety of flight structural components or regions of safety of flight structural components which are either sized by the requirements of this specification (Category I fracture critical parts), or could be sized by the requirements of this specification if fracture control procedures are not employed (Category II fracture critical parts).

6.2.7 Minimum assumed in-service damage size. The minimum assumed in-service damage size is the smallest damage which shall be assumed to exist in the structure after completion of an in-service inspection.

6.2.8 Slow crack growth structure. Slow crack growth structure consists of those design concepts where flaws or defects are not allowed to attain the critical size required for unstable rapid propagation. Safety is assured through slow crack growth for specified periods of usage depending upon the degree of inspectability. The strength of slow crack growth structure with subcritical damage present shall not be degraded below a specified limit for the period of unrepaired service usage.

6.2.9 Crack arrest fail safe structure. Crack arrest fail safe structure is structure designed and fabricated such that unstable rapid propagation will be stopped within a continuous area of the structure prior to complete failure. Safety is assured through slow crack growth of the remaining structure and detection of the damage at subsequent inspections. Strength of the remaining undamaged structure will not be degraded below a specified level for the specified period of unrepaired service usage.

6.2.10 Multiple load path-fail safe structure. Multiple load path fail safe structure is designed and fabricated in segments (with each segment consisting of one or more individual elements) whose function it is to contain localized damage and thus prevent complete loss of the structure. Safety is assured through slow crack growth in the remaining structure to the subsequent inspection. The strength and safety will not degrade below a specified level for a specified period of unrepaired service usage.

6.2.10.1 Multiple load path-dependent structure. Multiple load path structure is classified as dependent if, by design, a common source of cracking exists in adjacent load paths at one location due to the nature of the assembly or manufacturing procedures. An example of multiple load path-dependent structure is planked tension skin where individual members are spliced in the spanwise direction by common fasteners with common drilling and assembly operations.

6.2.10.2 Multiple load path-independent structure. Multiple load path structure is classified as independent if by design, it is unlikely that a common source of cracking exists in more than a single load path at one location due to the nature of assembly or manufacturing procedures.

6.3 Ordering data. MIL-A-008866A(USAF) dated 31 March 1971 and MIL-A-008867A(USAF) dated 31 March 1971 or later issue will be used in conjunction with this specification.

Custodian:
Air Force - 11

Preparing activity:
Air Force - 11

Project No. 1510-F022

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MIL-A-008866B(USAF)
22 August 1975
SUPERSEDING
MIL-A-008866A(USAF)
31 March 1971
USED IN LIEU OF
MIL-A-8866(ASG)
18 May 1960

MILITARY SPECIFICATION

AIRPLANE STRENGTH AND RIGIDITY RELIABILITY REQUIREMENTS, REPEATED LOADS AND FATIGUE

This limited coordination Military Specification has been prepared by the Air Force based upon currently available technical information, but it has not been approved for promulgation as a coordinated revision of Military Specification MIL-A-8866(ASG). It is subject to modification. However, pending its promulgation as a coordinated Military Specification, it may be used in procurement.

1. SCOPE

1.1 This specification identifies the durability design requirements applicable to the structure of airplanes. The complete structure, herein referred to as the airframe, includes the fuselage, wing, empennage, landing gears, control systems and surfaces, engine mounts, structural operating mechanisms, and other components as specified in the contract. This specification applies to metallic and nonmetallic structures. The objective is to minimize the in-service maintenance costs and to obtain operational readiness through proper controls on materials selection and processing, inspections, design details, stress levels, and protection systems.

2. APPLICABLE DOCUMENTS

2.1 The following documents of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein.

SPECIFICATIONS

Military

MIL-A-8861	Airplane Strength and Rigidity, Flight Loads
MIL-A-8867	Airplane Strength and Rigidity Ground Tests
MIL-A-8870	Airplane Strength and Rigidity, Flutter, Divergence, and Other Aeroelastic Instabilities
MIL-A-8871	Airplane Strength and Rigidity, Flight and Ground Operations Tests

FSC 1510

MIL-A-8892 Airplane Strength and Rigidity, Vibration
MIL-A-8893 Airplane Strength and Rigidity, Sonic Fatigue

(Copies of specifications, standards, drawings and publications required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

3. REQUIREMENTS

3.1 General requirements. The airframe shall be designed such that the economic life is greater than the design service life when subjected to the design service loads/environment spectra. The design objective is to minimize cracking or other structural or material degradation which could result in excessive maintenance problems or in functional problems such as fuel leakage, loss of control effectiveness or loss of cabin pressure. The contractor shall perform the analytical and experimental work necessary to demonstrate compliance with the analysis and tests as required herein, MIL-A-8867, and the contract. The design flight-by-flight load, stress, and environmental spectra shall be developed by the contractor. Spectra interaction effects such as that due to variable loading and environment shall be accounted for in the design.

3.2 Detail requirements

3.2.1 Design service life and design usage. The design service life and design usage will be specified by the procuring activity in the contract. The design service life and design usage will be based on the mission requirements and will be stipulated in terms of:

a. Total flight hours.

b. Total number of flights.

c. Total number and type of landings.

d. Total service years.

e. Mission profiles for each type of mission to be flown. (These profiles will be divided into mission segments such as taxi, takeoff run, a[illegible]nt, cruise, low altitude usage, inflight refueling, air-to-air combat, air-to-ground combat, etc. The mission profiles will also stipulate the approximate duration, altitude, speed, and payload configuration requirements for each mission segment.)

f. Mission mix or number of flights of each mission.

g. Any other special requirements such as functional check flights, ground maintenance operational checks, etc.

3.2.2 Design service loads spectra. The design service loads spectra for the airframe shall be developed for the design service life and typical design usage of 3.2.1 and shall require approval by the procuring activity. The contractor shall include all significant sources of repeated loads. The sources of repeated loads shall include, but not be limited to, engine ground run-up, functional check-outs, jacking, towing, taxiing, landing, flight maneuvers, atmospheric turbulence, inflight refueling, control system operation, cabin pressurization, buffeting, and terrain following maneuvers. The individual spectra of repeated loads for a particular airplane shall be based on the data referenced in the following subparagraphs as modified and amplified due to the existence of more representative data or unique airplane requirements. The individual spectra of repeated loads shall be assembled on a flight-by-flight basis to form the design service loads sequence. Load occurrences less than once per mission segment or once per flight shall be rationally distributed (randomized or ordered, as appropriate) among appropriate segments and flights. The design service loads spectra shall not be arbitrarily limited to design static limit load if higher values are probable (e.g., once per lifetime airplane load level). An appropriate distribution of weight, center of gravity, configuration, speed, altitude, and other significant operational parameters shall be made within each mission segment.

3.2.2.1 Maneuver. Tables I through VI contain normal maneuver load factor spectra representative of USAF operations of several classes of airplanes prior to 1970 and are contained herein for reference. The contractor shall derive the final maneuver spectra by mission segment and account for variables such as maneuver capability, tactics, etc. These final spectra shall require approval by the procuring activity.

3.2.2.2 Gust. The gust loads spectra shall be developed in accordance with the procedures of MIL-A-8861, paragraph entitled, Continuous turbulence analysis.

3.2.2.3 Landing. The landing loads spectra shall be developed for the number of landings indicated in 3.2.1 including such variables as sinking speed, forward speed, attitude, wing stores, and fuel distribution. The distribution of sinking speeds specified in table VII is for reference.

3.2.2.4 Taxi. The taxi ground loads shall be based on vertical gear inputs resulting from taxi on prepared runways. Table VIII is presented for reference. For airplane classes B_{II}, $C_{ASSAULT}$, and $C_{TRANSPORT}$, the number of vertical load cycles shall be twice that as specified in table VIII; or the taxi ground loads spectra shall account for the increased frequency of vertical gear inputs due to unprepared field operations. In lieu of a vertical load cycle spectra such as table VIII, the airplane may be designed for the effect of takeoff, taxiing, and rollout on deterministic runway profiles having power spectral density roughness characteristics as shown on figures 1, 2, and 3 or in the contract. For this option, the analysis shall include the significant rigid and flexible body modes, and gear dynamics. Aerodynamic and propulsion forces shall be included. The number of taxi operations for each of the runway roughness and airfield types shall be specified by the procuring activity, and taxi times and speeds shall require approval by the procuring activity.

3.2.2.5 Braking, pivoting, and turning. Taxi ground loads spectra shall include lateral and longitudinal loads resulting from braking, pivoting, and turning. Hard braking with maximum braking effects will be assumed to occur twice per full-stop landing and medium braking with half-maximum braking effects will be assumed to occur an additional five times per full-stop landing. During a given mission, each full-stop landing that occurs will be included. The effects of antiskid devices will also be included. Pivoting, with half-limit torque load, will be assumed to occur every 10 landings. Turning with a side load factor acting at the airplane center of gravity, reacted by the landing gears alternately inboard and outboard, will be assumed to occur. The magnitude and frequency of occurrences per landing of side load factor will be established by the contractor and will require approval by the procuring activity.

3.2.2.6 Pressurization. The number of pressurization cycles used for design shall be determined by, and be commensurate with, the design usage and design life requirements. Regulator valve nominal setting shall define the maximum pressure for cabins and cockpits.

3.2.2.7 Repeated operation of movable structures. Particular attention shall be given to the impact loads as well as the operational and residual loads that may occur when doors, cowling, landing gear, controls, and other devices are operated consistent with planned usage of the airplane.

3.2.2.8 Control surface balance weight attachments. Repeated load requirements for design of control surface balance weight attachments shall be in accordance with MIL-A-8870, paragraph entitled Mass-balance control surfaces and tabs [sub para (b).]

3.2.2.9 Control system inputs. The design service loads spectra shall include loads generated in performing the selected manual or automatic control functions. Rigid body and flexible modes of the airplane as well as the frequency response characteristics of the control system (including any filters used to modify response to structural modes) shall be considered in the derivation of the load spectra. The loads spectra due to pilot induced maneuvers shall be based on manual command inputs that are rationally derived.

3.2.2.10 Combined loadings. Loading conditions from individual sources of repeated loads shall be combined where appropriate. The contractor shall submit the rationale for combining individual sources of repeated loads to the procuring activity for approval.

3.2.3 Design chemical/thermal environment spectra. The contractor shall develop design chemical/thermal environment spectra. These spectra shall characterize each environment (i.e., intensity, duration, frequency of occurrence, etc.). The chemical/thermal environment spectra shall require approval by the procuring activity.

3.2.4 Analyses. An analysis shall be conducted to demonstrate that the economic life of the airframe is in excess of the design service life when subjected to the design service loads spectra and the design chemical/thermal environment spectra. The approach shall account for those factors affecting the time for cracks or other damage to reach sizes large enough to necessitate the repair, modification, or replacement of components. These factors shall include initial quality and initial quality variations, environment, load sequence and environment interaction effects, material property variations, and analytical uncertainties. The analysis shall demonstrate that cracks in the structure throughout one design lifetime shall not result in sustained crack growth under steady state flight (1G) and ground stress conditions. The design and analyses procedures shall be verified by test to selected design flight-by-flight stress and environment spectra and shall require approval by the procuring activity.

3.2.5 Durability detail design procedures. The contractor shall implement a disciplined procedure for durability design which will minimize the probability of incorporating adverse residual stresses, local design details, materials, processing and fabrication practices into the airplane design and manufacture which could lead to unexpected cracking or failure problems (i.e., those problems which have historically been found early in durability testing or early in service usage). The procedure shall be implemented concurrently with strength design. The procedure shall adequately reflect previous full scale test and fleet experiences as well as other laboratory and development test results. In addition, it shall encompass those managerial actions necessary to monitor and control the durability detail design activities.

3.2.6 Thermal protection. Where structural designs and thermal analyses defining temperature distributions, thermal strain histories, and material allowables are based on use of thermal protection systems (e.g., surface finishes, platings, primers, paints, fire retardant and insulating barriers, etc.), these systems shall be designed and demonstrated to endure the design environment spectra of 3.2.3 unless it can be shown that replacement, repair, or refurbishment at shorter intervals is cost effective. Designing to usage intervals less than the design service life shall require approval by the procuring activity, and the intervals shall not be less than the design inspection intervals specified in the contract.

3.2.7 Corrosion protection. Where structural designs and strength and durability analyses are based on the use of corrosion protection systems (e.g., corrosion resistant materials, ventilation, drainage, and chemically resistant finishes, coatings or barriers), these systems shall be designed and demonstrated to endure the design environment spectra of 3.2.3 unless it can be shown that replacement, repair, or refurbishment at shorter intervals is cost effective. Designing to usage intervals less than the design service life shall require approval by the procuring activity, and the intervals shall not be less than the design inspection intervals specified in the contract. The corrosion prevention and control plan shall be as specified in the contract.

3.2.8 Wear endurance. Excessive wear of structural components, elements, and major bearing surfaces which would interfere with function of the part shall not occur within the design service life and design usage unless it can be shown that replacement, repair, or refurbishment at shorter intervals is cost effective. Designing to usage intervals less than the design service life shall require approval by the procuring activity, and the intervals shall not be less than the design inspection intervals specified in the contract. Wear endurance during movement of structural surfaces and elements shall be considered as well as wear endurance of maintenance access panels, doors, and other removable parts during repeated removal, ground handling, and reinstallation.

3.2.9 Other durability considerations. The contractor shall develop and apply criteria for other durability considerations such as foreign object damage and special environments such as runway debris, sand, gravel, rain, hail, and lightning strikes. These considerations can arise due to airplane configurations, operation on substandard runways, or special atmospheric conditions. The criteria for these other durability considerations shall require approval by the procuring activity.

4. QUALITY ASSURANCE PROVISIONS

4.1 Design data. Structural design and analysis data shall be prepared and submitted as specified in the contract.

4.2 Laboratory tests. Laboratory tests shall be in accordance with MIL-A-8867.

4.3 Flight tests. Flight tests shall be in accordance with MIL-A-8871.

5. PREPARATION FOR DELIVERY

5.1 Section 5 is not applicable to this specification.

6. NOTES

6.1 Intended use. This specification is intended for use in the design of the airframe of all new USAF airplanes. Selected parts of this specification may also be used in the design of major modifications of existing USAF airplanes. Vibration and sonic durability design requirements are contained in MIL-A-8892 and MIL-A-8893, respectively.

6.2 Definitions. Definitions will be in accordance with the documents listed in Section 2 and as specified herein.

6.2.1 Durability. The ability of the airframe to resist cracking (including stress corrosion and hydrogen induced cracking), corrosion, thermal degradation, delamination, wear, and the effects of foreign object damage for a specified period of time.

6.2.2 Economic life. That operational life indicated by the results of the durability test program (i.e., test performance interpretation and evaluation in accordance with MIL-A-8867) to be available with the incorporation of USAF approved and committed production or retrofit changes and supporting application of the force structural maintenance plan in accordance with MIL-STD-1530. (In general, production or retrofit changes will be incorporated to correct local design and manufacturing deficiencies disclosed by the test. It will be assumed that the economic life of the test article has been attained with the occurrence of widespread damage which is uneconomical to repair and, if not repaired, could cause functional problems affecting operational readiness. This can generally be characterized by a rapid increase in the number of damage locations or repair costs as a function of cyclic test time.)

6.2.3 Initial quality. A measure of the condition of the airframe relative to flaws, defects, or other discrepancies in the basic materials or introduced during manufacture of the airframe.

6.2.4 Structural operating mechanisms. Those operating, articulating, and control mechanisms which transmit structural forces during actuation and movement of structural surfaces and elements.

6.3 Marginal indicia. Asterisks are not used in this revision to identify changes with respect to the previous issue due to the extensiveness of the changes.

Custodian:
Air Force - 11

Preparing activity
Air Force - 11

Project No. 1510-F023

TABLE I. Maneuver-Load-Factor Spectra for A, F, TF Classes, Cumulative Occurrences per 1000 Flight Hours by Mission Segment

N_z	Ascent	Cruise	Descent	Loiter	Air-Grnd	Spec Wpn	Air-Air
				Positive			
2.0	5000	10,000	20,000	15,000	175,000	70,000	300,000
3.0	90	2,500	5,500	2,200	100,000	25,000	150,000
4.0	1	400	500	250	40,000	7,500	50,000
5.0		1	1	25	10,000	2,000	13,000
6.0				1	1,500	250	3,300
7.0					200	15	900
8.0					15	1	220
9.0					1		60
10.0							15
				Negative			
0.5					10,000		44,000
0					350		4,000
-0.5					30		1,200
-1.0					7		350
-1.5					3		60
-2.0					1		8
-2.5							1

TABLE II. Maneuver-Load-Factor Spectra for T, Trainer Class, Cumulative Occurrences per 1000 Flight Hours by Mission Type

N_z	Transition		Formation	Instruments & Navigation	Administrative & Test
	Basic	Advanced			
Positive					
2.0	20,000	35,000	30,000	10,000	25,000
2.5	4,000	18,000	10,000	2,500	8,000
3.0	1,500	10,000	4,000	1,000	3,500
3.5	500	5,000	2,200	500	1,700
4.0	150	2,500	1,200	250	900
4.5	55	1,000	600	55	450
5.0	20	350	250	20	170
5.5	7	110	90	7	50
6.0	3	30	25	3	15
6.5		9	5	1	4
7.0		2.5	1		
7.5		1			
Negative					
0	280	3,700	4,800	1,200	2,800
-0.5	38	320	160	38	330
-1.0	20	100	20	3	110
-1.5	4	34	6	1	47
-2.0		10	1	0.6	18
-2.5		2			

TABLE III. Maneuver-Load-Factor Spectra for B_I, Bomber Class, Cumulative Occurrences per 1000 Flight Hours by Mission Type

N_z	Special Weapons	Transition	Air-Ground
		Positive	
1.3	29,000	10,000	31,000
1.6	22,500	5,000	29,000
1.9	17,500	2,700	20,000
2.2	12,500	1,300	15,000
2.5	8,000	600	10,000
2.8	5,000	275	6,500
3.1	2,800	100	4,000
3.4	1,600	45	2,400
3.7	800	18	1,[illegible]50
4.0	400	7	650
4.3	200	2	300
4.6	70	1	130
5.0	15		35
		Negative	
0.7	525	250	1,300
0.5	450	150	1,000
0.3	350	85	700
0.1	235	35	450
0	170	17	350
-0.1	100	7	250
-0.3	1	1	90
-0.5			1

TABLE IV. Maneuver-Load-Factor Spectra for B_{II}, Bomber Class Cumulative Occurrences per 1000 Flight Hours by Mission Segment

N_z	Ascent	Cruise	Descent	Refueling
Positive				
1.2	54,000	13,000	50,000	40,000
1.4	7,200	1,300	6,000	1,300
1.6	900	150	600	100
1.8	100	25	80	10
2.0	10	4	15	1
2.2	1	1	4	
2.4		0.15	1	
2.6		0.08	0.3	
2.8			0.1	
3.0			0.03	
Negative				
0.9	80,000	20,000	85,000	260,000
0.8	26,000	4,200	31,000	30,000
0.7	7,700	960	11,000	4,300
0.6	2,500	240	3,900	830
0.5	780	60	1,050	200
0.3	86	4	51	20
0.1	16	0.7	4	3.5
0	7		0.7	1
-0.2	1.5			
-0.4	0.4			
-0.6	0.1			

TABLE V. Maneuver-Load-Factor Spectra for $C_{TRANSPORT}$, Cargo Class, Cumulative Occurrences per 1000 Flight Hours by Mission Segment

N_z	Logistics			Training			Refuel
	Ascent	Cruise	Descent	Ascent	Cruise	Descent	
Positive							
1.2	11,000	825	13,000	60,000	45,000	35,000	8,000
1.4	380	30	435	5,600	4,000	3,500	850
1.6	25	3	28	500	350	800	110
1.8	4.5	0.7	5	70	35	250	20
2.0	1.8			15	5	90	2.5
2.2				4	1	35	
2.4				2		11	
2.6				1		4.5	
2.8						1.5	
Negative							
0.9	6,800	600	7,000	12,000	7,200	10,000	3,000
0.8	2,500	150	3,000	5,000	1,500	1,700	800
0.7	600	75	680	1,000	200	350	200
0.6	100	20	120	200	30	85	70
0.4	1	0.8	1	7	1	7	8
0.2						0.6	2

TABLE VI. Maneuver-Load-Factor Spectra for $C_{ASSAULT}$, Cargo Class, Cumulative Occurrences per 1000 Flight Hours by Mission Segment

N_z	Ascent	Cruise	Descent
		Positive	
1.2	14,000	3,500	26,000
1.4	1,000	300	1,500
1.6	120	35	300
1.8	15	6	50
2.0	3	2	10
2.2	0.7	0.7	3
2.4		0.4	1
2.6		0.25	0.5
2.8			0.3
3.0			0.18
3.2			0.12
		Negative	
0.8	4,700	1,000	5,000
0.6	105	30	100
0.4	8	3	3
0.2	2	0.3	
0	0.5		

TABLE VII. Cumulative Occurrences of Sinking Speed/1000 Landings

Sinking Speed FPS	Trainer	All Other Classes
0.5	1000	1000
1.5	870	820
2.5	680	530
3.5	460	270
4.5	270	115
5.5	145	37
6.5	68	11
7.5	31	3.0
8.5	14	1.5
9.5	6.0	0.5
10.5	3.0	0
11.5	1.5	
12.5	0.5	
13.5	0	

TABLE VIII. Cumulative Occurrences Per Thousand Runway Landings That Load Factor N_z is Experienced at the Airplane CG

N_z	Cumulative Occurrences
1 ±0	494,000
1 ±0.1	194,000
1 ±0.2	29,000
1 ±0.3	2,100
1 ±0.4	94
1 ±0.5	4
1 ±0.6	0.155
1 ±0.7	0.005
1 ±0.8	0

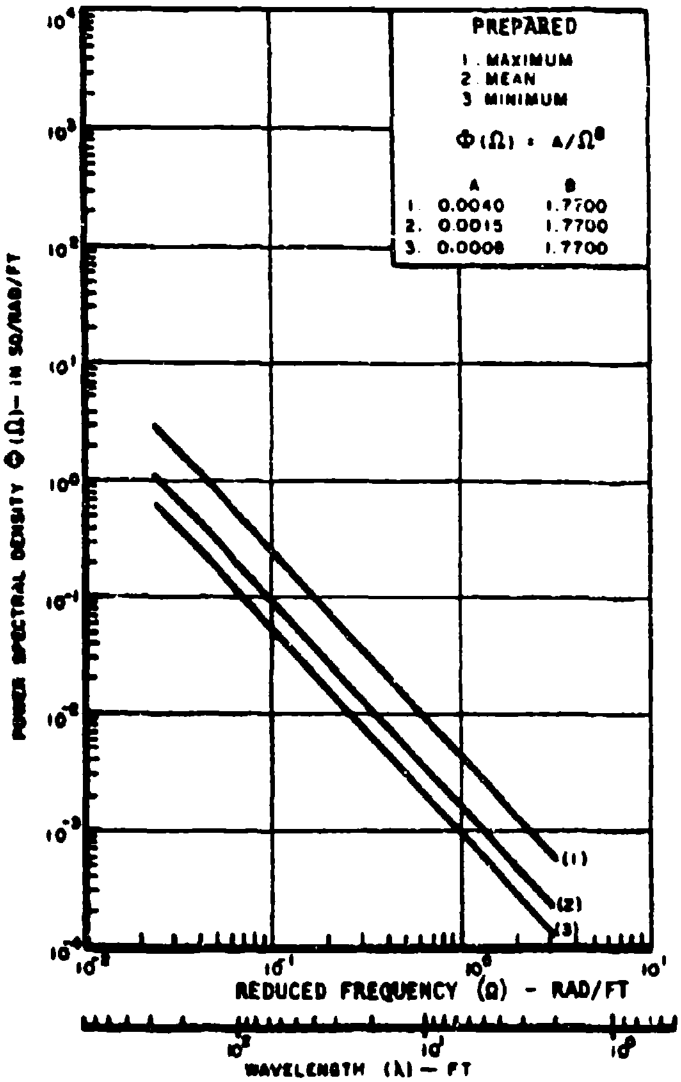

FIGURE 1. Roughness Levels For Prepared Airfields

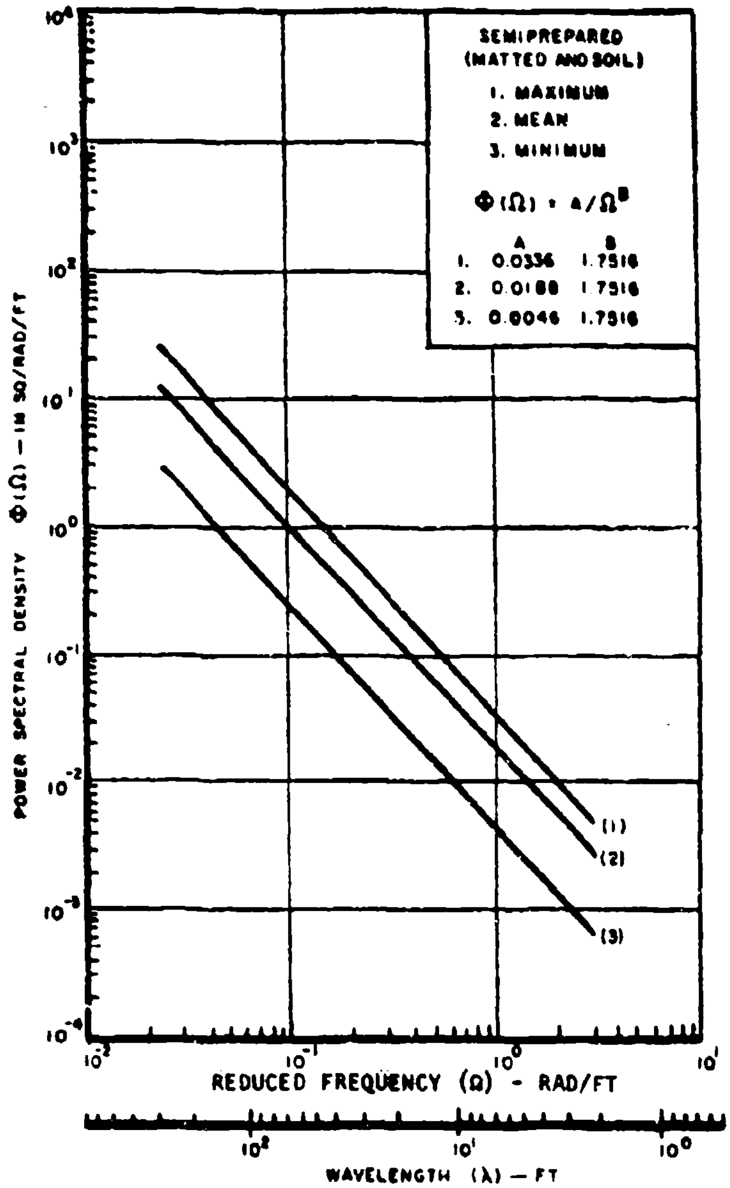

FIGURE 2. Roughness Levels For Semiprepared Airfields

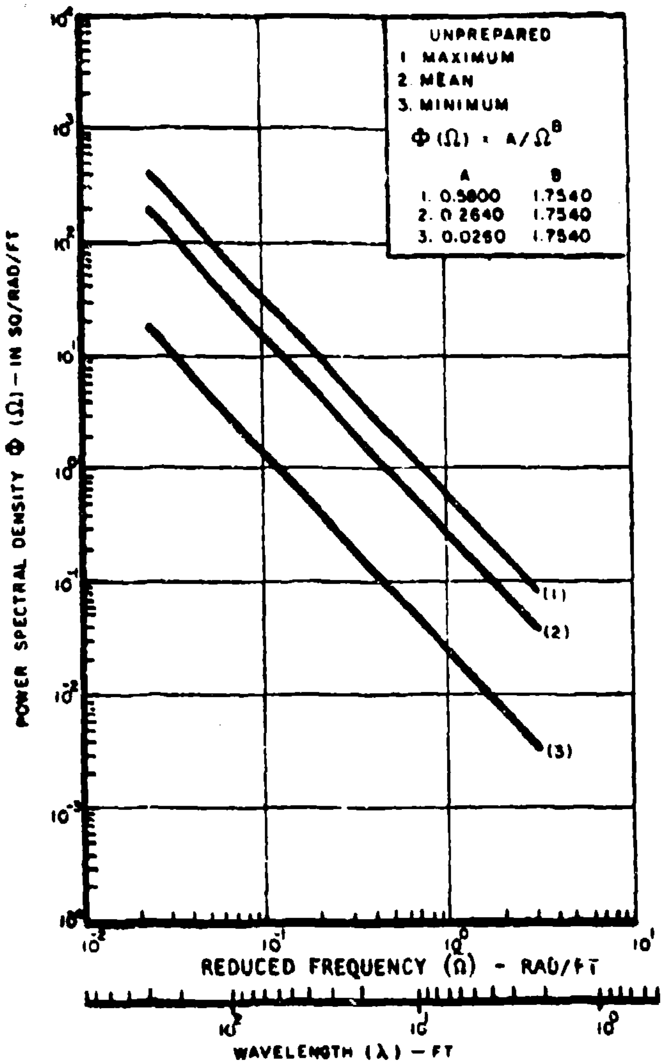

FIGURE 3. Roughness Levels For Unprepared Airfields [1]

NOTE 1: This figure shall not be used for design unless specified by the procuring activity.

ALPHABETICAL INDEX

ALPHABETICAL INDEX (CONT'D)

ALPHABETICAL INDEX (Cont'd)

TABLES

FIGURES

★U.S. GOVERNMENT PRINTING OFFICE: 1978-603-761/900

STANDARDIZATION DOCUMENT IMPROVEMENT PROPOSAL	OMB Approval No. 22-R255

INSTRUCTIONS: The purpose of this form is to solicit beneficial comments which will help achieve procurement of suitable products at reasonable cost and minimum delay, or will otherwise enhance use of the document. DoD contractors, government activities, or manufacturers/vendors who are prospective suppliers of the product are invited to submit comments to the government. Fold on lines on reverse side, staple in corner, and send to preparing activity. Comments submitted on this form do not constitute or imply authorization to waive any portion of the referenced document(s) or to amend contractual requirements. Attach any pertinent data which may be of use in improving this document. If there are additional papers, attach to form and place both in an envelope addressed to preparing activity.

DOCUMENT IDENTIFIER AND TITLE

NAME OF ORGANIZATION AND ADDRESS	CONTRACT NUMBER
	MATERIAL PROCURED UNDER A ☐ DIRECT GOVERNMENT CONTRACT ☐ SUBCONTRACT

1. HAS ANY PART OF THE DOCUMENT CREATED PROBLEMS OR REQUIRED INTERPRETATION IN PROCUREMENT USE?
   A. GIVE PARAGRAPH NUMBER AND WORDING.

   B. RECOMMENDATIONS FOR CORRECTING THE DEFICIENCIES

2. COMMENTS ON ANY DOCUMENT REQUIREMENT CONSIDERED TOO RIGID

3. IS THE DOCUMENT RESTRICTIVE?

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4. REMARKS

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	DATE

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MIL-A-008867B(USAF)
22 August 1975
SUPERSEDING
MIL-A-008867A(USAF)
31 March 1971
USED IN LIEU OF
MIL-A-8867(ASG)
18 May 1960

MILITARY SPECIFICATION

AIRPLANE STRENGTH AND RIGIDITY GROUND TESTS

This limited coordination Military Specification has been prepared by the Air Force based upon currently available technical information, but it has not been approved for promulgation as a coordinated revision of Military Specification MIL-A-8867(ASC). It is subject to modification. However, pending its promulgation as a coordinated Military Specification, it may be used in procurement.

1. SCOPE

1.1 Types of tests. This specification identifies the ground tests required for structural evaluation of airplanes. The complete structure, herein referred to as the airframe, includes the fuselage, wing, empennage, landing gears, control system and surfaces, engine mounts, structural operating mechanisms, and other components as specified in the contract. This specification applies to metallic and nonmetallic structures. The types of testing include, but are not limited to:

a. Design development tests

b. Proof, ultimate, and failing load static tests - full scale airframe

c. Durability tests - full scale airframe

d. Damage tolerance tests

e. Fuel tank tests.

2. APPLICABLE DOCUMENTS

2.1 The following documents, of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein:

SPECIFICATIONS

Military

MIL-G-6021 Castings, Classification and Inspection of

FSC 1510

MIL-A-8860	Airplane Strength and Rigidity, General Specification for
MIL-A-8866	Airplane Strength and Rigidity Reliability Requirements, Repeated Loads, and Fatigue
MIL-A-8871	Airplane Strength and Rigidity, Flight and Ground Operations Tests
MIL-C-45662	Calibration Systems Requirements
MIL-A-83444	Airplane Damage Tolerance Design Requirements

(Copies of specifications, standards, drawings and publications required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

3. REQUIREMENTS

3.1 General requirements. The contractor shall furnish component, assembly, and full scale airframe test specimens and shall perform tests in accordance with the test requirements specified herein and as modified and amplified by the contract.

3.1.1 Location of tests. The contract will specify whether the tests are to be performed by the Government or by the contractor. In the event that structural tests are performed by the Government, the contract will specify the type and amount of support to be provided by the contractor.

3.1.2 Schedule of tests. The test scheduling shall be as specified in the detail requirements of 3.2. In all cases, the test sequencing shall require approval by the procuring activity prior to starting the test program.

3.1.3 Test articles. Test article configuration shall require approval by the procuring activity. Changes, adjustments, reinforcements and repairs made to the test article to meet specified strength, rigidity, damage tolerance and durability requirements shall be representative of those that will be incorporated into operational flight articles. In addition, the test articles shall be identical with the structure of the flight articles except that:

a. Items such as fixed equipment and useful loads and their support structures may be omitted from the test structure provided the omission of these parts does not significantly affect the load, stress or thermal distributions and the structural characteristics of the parts of the structure to be tested, and provided the omitted parts are qualified by separate tests as agreed to by the procuring activity.

b. Substitute parts may be used when specific prior approval is obtained from the procuring activity, provided they produce the effects of the parts for which they are substituted, and provided the structural integrity of the parts for which substitutions are made are demonstrated in a manner that is satisfactory to the procuring activity.

c. Power plants and accessories shall be replaced by contractor designed-and-fabricated test fixtures that properly transmit the power plant loads to the engine mounts, vibration isolators, or both, as applicable. The means for applying the loads to these fixtures (such as loading rods through the fuselage or engine nacelle structure) shall be determined by the contractor. All structural modifications necessary to accommodate the loading devices shall be designed by the contractor in such a manner as to assure that the structural characteristics of the modified structure will be equivalent to those of the actual structure.

d. Paint or other finishes that do not affect the structural performance may be omitted from the test structures. When the structural test includes simulation of chemical or thermal environment (3.1.9), the test articles shall include the associated environmental protection systems developed in accordance with the durability design requirements of MIL-A-8866 paragraphs entitled Thermal protection and Corrosion protection.

e. Prior to shipping the test structures to Government facilities for testing, a number of buttock lines, water lines, fuselage stations, and wing stations shall be marked on the test structure. These shall be clearly identified and shall be of sufficient number to facilitate determining all desired reference points on the airframe.

f. To the extent required for adequate load simulation during test, mechanical portions of the flight control system and power actuators for the control systems shall be operable. When tests are conducted at Government facilities, special provisions shall be made for external power attachments to the actuating mechanisms to permit externally controlled operations. It is therefore permissible to omit any unnecessary portions of the normal internal power systems. Other actuators for landing gear doors, armament bay doors, etc., shall be externally operable as required for tests at Government facilities. Air actuated systems may be replaced by hydraulic systems to simplify testing procedures. The external actuation capability is also recommended for tests conducted by the contractor, if test operations can be simplified or costs reduced.

g. Structural parts and mechanisms which are subject to special qualification requirements outside the scope of this specification shall be qualified to the extent possible prior to incorporation in the test article (Class I castings in accordance with MIL-C-6021 paragraph entitled, Classes, weldments, actuators, etc.).

3.1.4 Instrumentation and test measurements. Structural test components, assemblies, and full scale airframe test articles shall be instrumented with strain gages, load cells, pressure transducers, deflection potentiometers, thermocouples, and other instrumentation as needed to (1) verify that external loads, pressure loads, environment and other external test parameters are correctly simulated and (2) monitor test article parameters such as strain, temperature distributions, and structural deflections for comparison with the appropriate structural analyses. Additional instrumentation shall be used as necessary to detect incipient structural failure, monitor crack growth, and monitor localized test areas. The instrumentation system sensor placement on the structural test articles shall be determined by the contractor and shall require approval by the procuring activity. Instrumentation used for obtaining test data shall be calibrated and certified in accordance with MIL-C-45662 as appropriate. Test facility measurement standards shall have certificates which are traceable to the National Bureau of Standards. The instrumentation shall be integrated into a readout system for rapid and accurate presentation of the test parameters. Data measurements shall be taken at sufficient intervals and loadings to monitor and verify the test parameters consistent with the test program objectives. For tests conducted at discrete load increments, measurements shall be made at each load increment. The test article instrumentation requirements shall be coordinated with the instrumentation planned for the flight loads survey. Special types of instrumentation (e.g., mechanical strain recorders, strain gages, etc.) to be used during the individual airplane tracking program shall be placed on the static and durability articles as appropriate for evaluation and correlation. Analyses pertinent to the areas being instrumented shall be made available to the procuring activity prior to instrumentation. When tests are performed by the Government, required instrumentation (strain gages, thermocouples, pressure transducers, crack detection wires, etc.) shall be installed by the contractor to the maximum extent practicable, prior to delivery of the test article(s) to the testing agency. When tests are performed by the Government, the contractor shall consult with the testing agency to establish the instrumentation requirements relative to compatibility with Government data systems.

3.1.5 Use and disposition of test articles. Except for the case of proof testing of flight vehicle structures, parts of the test structure shall not be used on a flight article. In certain cases it may be a program requirement to store test articles for extended periods of time following completion of testing. The requirements for test article storage shall be as specified by the procuring activity in the contract.

3.1.6 Test loading system. The test loads shall be applied using a system capable of providing accurate load control to all load points simultaneously and shall contain emergency modes which will detect load errors and prevent excessive loads. When loads are applied in such a manner that they are not relieved when the rate of deformation of the specimen increases rapidly, as when failure occurs, safety devices such as shear links or pressure blowout valves shall be employed to minimize excessive deformation or overloading of other parts of the structure. Positive methods shall be employed to safely control the release of energy in the event of abrupt failure. Load application devices shall be designed to minimize local non-representative loading effects and to afford maximum accessibility for inspection of critical joints, cutouts, and areas of discontinuity. The test rig and associated equipment shall be capable of applying the maximum loads necessary to meet the required test objectives.

3.1.7 Test loads and distribution. In each test condition, parts of the structure critical for the pertinent design loading shall be tested and shall be loaded simultaneously, if practicable. Testing may be initiated using analytically derived loads and available wind tunnel data. Loads measured in the flight and ground loads survey program shall be used to correct the test loads and distribution at the earliest suitable time if the measured loads are significantly different than the analytical loads. The distribution of loads employed in the tests shall represent the actual distribution as closely as possible.

3.1.8 Deformations. It shall be demonstrated during structural tests that movable and removable structural components remain in their intended positions and do not deform within the load/deformation limits specified in MIL-A-8860 paragraph entitled, Deformations, to the extent that (1) deleterious aerodynamic effects are produced or (2) interference is such that functional impairment occurs when operation is required at the design condition. In addition, there shall be no permanent deformation as a result of application of the design loads specified in MIL-A-8860 paragraph entitled, Deformations which would impair the functioning of any aircraft component during subsequent flight and ground operations.

3.1.9 Environmental effects. The effect of chemical and thermal environments shall be evaluated during the material and joint allowables tests to the extent necessary. When deemed necessary, the design chemical and thermal environment shall be simulated during the full scale airframe tests. The method of simulating the environment shall require approval by the procuring activity.

3.1.10 Simplification and combination of loading. Loading conditions may be simplified during tests by modifying the distribution of loads applied to regions of a structure that will not be subjected to critical loads during the loading condition being simulated or that are identical in construction to other regions of the structure that are subjected to critical loads during the same or another test condition. However, simplification of the method of loading shall not result in unrepresentative permanent deformations or failures. Simultaneously applying more than one loading condition to different portions of the structure shall be considered provided the interaction of the separate loadings does not affect the critical design loading on any portion of the structure. Loads resulting from pressurization shall be considered and, if critical, shall be simulated in combination with the applicable ground and flight loads during the appropriate component or full scale test.

3.1.11 Complete airframe versus separate assemblies. It will be a program option requiring approval by the procuring activity whether the full scale airframe static and durability tests are performed on a complete airframe or on separate major assemblies thereof (wing, fuselage, empennage, landing gear, etc.). When tests of components or separate assemblies are conducted, the test article shall be mounted in supporting and loading fixtures which accurately simulate the load and deflection interactions with the adjacent structure not being tested. If these actual interactions cannot be obtained, then the contractor shall provide sufficient transition test structure whose strength and stiffness is representative of the full scale airframe.

3.2 Detail requirements

3.2.1 Design development tests. The contractor shall conduct design development tests to establish material and joint allowables; to verify analysis procedures; to obtain early evaluation of allowable stress levels, material selections, fastener systems and the effect of the design chemical/thermal environment spectra; and to obtain early evaluation of the strength, durability, and damage tolerance of critical structural components and assemblies. Example of design development tests are tests of:

a. Coupons

b. Small elements

c. Splices and joints

d. Panels of basic section and panels with joints, cutouts, eccentricities and other discontinuities.

e. Fittings

f. Con .ol system components and structural operating mechanisms.

In addition, design development tests shall include tests of critical major components and assemblies such as wing carry through, horizontal tail spindles, wing pivots, and assemblies thereof to obtain early validation of the static strength, durability, and damage tolerance. A design development test plan shall be developed by the contractor and shall require approval by the procuring activity.

3.2.2 Static tests - full scale airframe. The static test airframe shall meet the applicable general requirements of 3.1. Full scale static tests to design ultimate loads shall be required except (1) where it is shown that the airframe and its loading are substantially the same as that used on previous aircraft where the airframe has been verified by full scale tests or (2) where the strength margins (particularly for stability critical structure) have been demonstrated by major assembly tests. When full scale ultimate load static tests are not performed, it shall be a program requirement to conduct a strength demonstration proof test in accordance with 3.2.2.4. Deletion of the full scale ultimate load static tests shall require approval by the procuring activity. Prior to starting the static tests, structural modifications required as a result of failures that occur during design development tests shall be incorporated into the test article or qualified by separate tests as agreed to by the procuring activity.

3.2.2.1 Schedule. The full scale static tests shall be scheduled such that the tests are completed in sufficient time to allow removal of the 80 percent limit restrictions on the flight test airplanes in accordance with MIL-A-8871 paragraph entitled, Operating limitations, and allow unrestricted flight within the design envelope on schedule.

3.2.2.2 Functional proof tests prior to first flight. Proof testing requirements prior to first flight for major flight control systems and surfaces, and major operating mechanisms (e.g., wing sweep, droopnose, etc.) shall be established on an individual basis for each new airplane. The purpose of these tests is to demonstrate that systems and mechanisms function satisfactorily when subjected to the applicable maximum operating loads. These tests may be performed with the associated load induced deflection in the movable surface and the airframe to which the movable surface is attached, and may be performed on suitable components when approved ty the procuring activity. Pressurized compartments shall be tested to 1.33 times maximum operating pressure (regulator valve nominal setting plus tolerance) on a flight article prior to pressurized flight. Each subsequent airplane shall be tested to at least 1.0 times the maximum operating pressure.

3.2.2.3 Inspection proof tests. Upon approval by the procuring activity and in conformance to MIL-A-83444, paragraph entitled, Initial flaw size; subparagraph Slow crack growth structure, the contractor may perform component, assembly, or complete airframe inspection proof tests on every airplane for the purpose of defining maximum possible initial flaw sizes or other damage when design constraints make the use of conventional Non-Destructive Inspection impractical or not cost effective.

3.2.2.4 Strength demonstration proof tests. Strength demonstration proof tests shall be conducted when design ultimate load static tests are not required. The proof test load levels shall be equal to or greater than the maximum loads contained in the design service loads spectra and in no cases shall the load levels be less than design limit load. The structure shall be loaded during the strength demonstration proof tests in accordance with 3.1.7. Test conditions shall be selected which substantiate the design limit envelope for each component of the airframe. The internal loads and stress analysis shall be used as a guide in determining the most critical load conditions. The contractor shall submit a list of recommended test conditions including the basis for selection. Re-proof tests shall be required when flight test data confirms that actual load distributions are more severe than those used in design. Strength demonstration proof tests and re-proof test requirements shall require approval of the procuring activity.

3.2.2.4.1 Post proof test inspection and analysis requirements. A post proof test inspection program shall be conducted. Special emphasis shall be placed on determining if detrimental deformations (3.1.8) have occurred in the airframe that would prevent the use of any structural part on a flight vehicle. The analysis program shall include extensive examination of instrumentation data to determine whether extrapolated ultimate internal stresses are above predicted values to the extent that flight restrictions or modifications are required. The specific inspection and analysis program shall require approval by the procuring activity.

3.2.2.5 Ultimate load tests. In accordance with 3.2.2, the static test program shall include tests to design ultimate load on the full scale static test airframe to verify the static ultimate strength of the airframe. Design ultimate load test conditions shall be selected which substantiate the design envelope for each component of the airframe. The internal loads and stress analysis shall be used as a guide in determining the most critical load conditions. The contractor shall submit a list of recommended test conditions to the procuring activity for approval.

3.2.2.6 Failing load tests. When ultimate load static tests are conducted, consideration shall be given to conducting failing load tests at the end of the static test program to substantiate special capabilities such as growth potential or emergency operations. Failing load tests shall be specified in the contract unless other uses of the article are specified in the contract.

3.2.3 Durability tests - full scale airframe. The durability test article shall meet the applicable general requirements of 3.1. Prior to starting the durability tests, structural modifications required as a result of failures that occur during design development tests shall be incorporated into the test article or qualified by separate tests as agreed to by the procuring activity.

3.2.3.1 Selection of test articles. The test article shall be an early Full Scale Development (FSD) or Research Development Test and Evaluation (RDT&E) airframe to meet the scheduling requirements of 3.2.3.2. This article shall be as representative of the operational configuration as practical within the schedule constraints. If there are significant design, material, or manufacturing changes between the test article and production airplanes, durability test of an additional article or selected components and assemblies thereof shall be required. The contractor in conjunction with the procuring activity shall identify additional test requirements and these additional tests shall require separate contract negotiations.

3.2.3.2 Schedule requirements. The full scale airframe durability test shall be scheduled such that one lifetime of durability testing plus an inspection of critical structural areas in accordance with 3.2.3.4.1 and 3.2.3.4.2 shall be completed prior to full production go ahead decision. Two lifetimes of durability testing plus an inspection of critical structural areas in accordance with 3.2.3.4.1 and 3.2.3.4.2 shall be scheduled to be completed prior to delivery of the first production airplane. If the economic life of the test article is reached prior to two lifetimes of durability testing, sufficient inspection in accordance with 3.2.3.4.1 and 3.2.3.4.2 and data evaluation shall be completed prior to delivery of the first production airplane to estimate the extent of required production and retrofit changes. In the event the original schedule for the production decision and production delivery milestones becomes incompatible with the above schedule requirements, a study shall be conducted to access the technical risks and cost impacts of changing these milestones.

3.2.3.3 Test spectra. The test spectra shall be based on the design service loads spectra and the design chemical/thermal environment spectra. The test spectra shall include rationally distributed missions, positive and negative loads (ordered or randomized, as appropriate), and shall be applied to the test article on a flight-by-flight basis. Test loads shall include significant sources of repeated loads and these loads shall be combined in the appropriate sequence. Chemical and thermal environment shall be included in accordance with 3.1.9. The test load and environment spectra shall require approval by the procuring activity.

3.2.3.3.1 Test spectra truncation. Truncation, elimination or substitution of load cycles in the test spectra to reduce test time and cost will be allowed. The contractor shall define by analysis and laboratory experiment the effect of the difference between the design spectra and the proposed test spectra on the time to reach detrimental crack sizes per the durability and damage tolerance requirements of MIL-A-8866 paragraph entitled General requirements and MIL-A-83444 paragraph entitled General requirements, respectively. The results of these analysis and experiments shall be used to establish the final test spectra and, as necessary, to interpret the test results. The final test spectra shall require approval by the procuring activity.

3.2.3.4 Inspections. Major inspection programs shall be conducted as an integral part of the full scale airframe durability test program. The inspection programs shall require approval by the procuring activity.

3.2.3.4.1 Design inspections. In-service inspections developed in accordance with the requirements of MIL-A-83444 paragraph entitled General requirements and the requirements of MIL-A-8866 paragraph entitled General requirements shall be programmed and conducted at the specified intervals and at the end of the test prior to the teardown inspection of 3.2.3.4.3. The inspection procedures shall be consistent with those proposed for use on force airplanes at the design inspection interval specified in the contract and shall account for the fact that accessibility to the test airframe may be different than for the flight configuration.

3.2.3.4.2 Special inspections. The contractor and procuring activity shall define special inspections (both type and interval) to monitor and status of critical areas identified during design, detecting additional critical areas not previously identified, and monitoring crack growth rates. These inspections shall be conducted at intervals as agreed to by the procuring activity and shall include the following intervals necessary to support the schedule requirements of 3.2.3.2: (1) at the end of one lifetime of test and (2) at the end of two lifetimes of test, or when the economic life of the test article is reached but prior to the teardown inspection of 3.2.3.4.3.

3.2.3.4.3 Teardown inspection. At the end of the full scale durability test including any scheduled damage tolerance tests, a destructive teardown inspection program shall be conducted. This inspection shall include disassembly and laboratory-type inspection of those critical structural areas identified in design as well as additional critical structure detected during the design and special inspections and during close visual examination while performing the disassembly. Fractographic examinations shall be conducted to obtain crack growth data and to assist in the assessment of the initial quality of the airframe and the degree of compliance with the durability requirements of MIL-A-8866 paragraph entitled General requirements and the damage tolerance requirements of MIL-A-83444 paragraph entitled General requirements.

3.2.3.5 Test duration. A minimum of two lifetimes of durability testing shall be conducted except when the economic life is reached prior to two lifetimes. If the economic life is reached prior to two lifetimes, the durability test shall be terminated and a decision made to perform either the teardown inspection or perform damage tolerance tests as required by 3.2.4 followed by the teardown inspection. If, at the end of two lifetimes, the economic life is not reached, a decision shall be made to (1) terminate durability testing and perform the teardown inspection, or (2) terminate the durability testing and perform damage tolerance testing followed by the teardown inspection, or (3) continue durability testing for an approved extended duration followed by either (1) or (2). At each of the above decision points the contractor shall submit his recommended course of action together with the rationale supporting this recommendation to the procuring activity for approval. As a minimum, the rationale for continuing durability testing beyond two lifetimes shall be based on (1) effects of possible usage extremes on life, (2) possible force life extension needs, (3) development and production schedules, and (4) magnitude of cracking problems encountered in two lifetimes of testing.

3.2.4 Damage tolerance tests. Damage tolerance tests shall be conducted to demonstrate compliance with the design requirements of MIL-A-83444 paragraph entitled General requirements. The type and quantity of tests depend on the design concepts and the number of fracture critical areas. The types of tests shall include crack growth evaluation of slow crack growth and fail safe structure as well as residual strength and life tests of fail safe structure subsequent to load path failure or crack arrest. The amount of full scale damage tolerance testing that is conducted is also dependent upon the extent that damage tolerance is demonstrated during the design development or full scale durability tests (i.e., number of cracking incidents and subsequent crack growth). The damage tolerance test program shall be of sufficient scope to verify Category I fracture critical parts in accordance with MIL-A-83444 paragraph entitled Fracture critical structure. Deletion of verification of certain fracture critical areas can be proposed based on similarity of materials and structural configurations and demonstrated knowledge of the applied stresses. The intent shall be to conduct damage tolerance tests on existing test hardware. This may include use of components and assemblies of the design development tests as well as the full scale static and durability test articles. Fracture critical areas of existing test hardware shall be evaluated to determine the nature of physical changes caused by previous testing to insure validity of damage tolerance tests. When necessary, additional structural components and assemblies shall be fabricated and tested to verify compliance with the damage tolerance requirements of MIL-A-83444 paragraph entitled General requirements. Detail test requirements (type of tests, proposed deletions, quantity, choice of specimens, pre-crack locations, etc.) shall be proposed by the contractor and shall require approval by the procuring activity.

3.2.5 Fuel tank tests. The internal fuel tanks critical for repeated loads due to pressure, inertia, fluid acceleration heads, vibration, or other flight and ground loads shall be tested. When the critical stress conditions cannot be reasonably simulated in the durability test or other required test programs, durability tests shall be conducted on full scale representative tank sections as approved by the procuring activity. The contractor shall propose, for approval by the procuring activity, a plan to detect cracks, delaminations, or other material failures that would cause fuel leaks throughout the test duration. The test duration for fuel tank tests shall be as specified in 3.2.3.5. If the fuel tank is a Category I fracture critical part, it shall require damage tolerance tests as specified in 3.2.4. These test requirements do not supersede other test requirements for evaluation of fuel tanks (slosh and vibration).

3.2.6 Interpretation and evaluation of test results. Each structural problem (failure, cracking, yielding, etc.) that occurs during the tests required by this specification shall be analyzed by the contractor to determine the cause, corrective actions, force implications, and estimated costs. The scope and interrelations of the various tasks within the interpretation and evaluation effort are illustrated in figure 1. The results of this evaluation shall demonstrate that the strength, rigidity, damage tolerance and durability design requirements are met. Structural modifications or changes derived from the results of the full scale tests to meet the specified strength, rigidity, damage tolerance, and durability design requirements shall be substantiated by subsequent tests of components, assemblies, or full scale article as appropriate. The test duration for durability modifications shall be as specified in 3.2.3.5. The contractor shall propose these additional test requirements together with the associated rationale to the procuring activity for approval.

4. QUALITY ASSURANCE PROVISIONS

4.1 Additional tests. If the tests specified herein and performed by the contractor are inadequate to prove that the test structure meets the specified requirements, the contractor or the procuring activity will propose amendments to the contract to include additional tests.

4.2 Test witnesses. Before performing a required test, the procuring activity shall be notified in sufficient time so that a representative may witness the test and certify results and observations. The procuring activity shall be informed if the test is such that interpretation of the behavior of the structure under load is likely to require engineering knowledge and experience so that a qualified engineer may witness the test and certify the observations and results recorded during the test.

4.3 Test data. Structural test data shall be prepared and submitted as specified in the contract.

5. PREPARATION FOR DELIVERY

5.1 Section 5 is not applicable.

6. NOTES

6.1 Intended use. This specification is intended to be used in conjunction with MIL-T-6053, MIL-A-8870, MIL-A-8871, MIL-A-8892 and MIL-A-8893 for the structural substantiation of airframe structure of all new USAF airplanes. Selected portions of this specification may also be used in the substantiation of major modifications of existing USAF airplanes.

6.2 Definitions. Definitions will be in accordance with the documents listed in Section 2 and as specified herein.

3.2.1 Durability. The ability of the airframe to resist cracking (including stress corrosion and hydrogen induced cracking), corrosion, thermal degradation, delamination, wear, and the effects of foreign object damage for a specified period of time.

6.2.2 Economic life. That operational life indicated by the results of the durability test program (i.e., test performance interpretation and evaluation in accordance with MIL-A-8867) to be available with the incorporation of USAF approved and committed production or retrofit changes and supporting application of the force structural maintenance plan in accordance with MIL-STD-1530. (In general, production or retrofit changes will be incorporated to correct local design and manufacturing deficiencies disclosed by the test. It will be assumed that the economic life of the test article has been attained with the occurrence of widespread damage which is uneconomical to repair and, if not repaired, could cause functional problems affecting operational readiness. This can generally be characterized by a rapid increase in the number of damage locations or repair costs as a function of cyclic test time.)

6.2.3 Initial quality. A measure of the condition of the airframe relative to flaws, defects or other descrepancies in the basic materials or introduced during manufacture of the airframe.

6.2.4 Structural operating mechanisms. Those operating, articulating, and control mechanisms which transmit structural forces during actuation and movement of structural surfaces and elements.

6.2.5 Damage tolerance. The ability of the airframe to resist failure due to the presence of flaws, cracks, or other damage for a specified period of unrepaired service usage.

6.2.6 Marginal indicia. Asterisks are not used in this revision to identify changes with respect to the previous issue due to the extensiveness of the changes.

Custodian:
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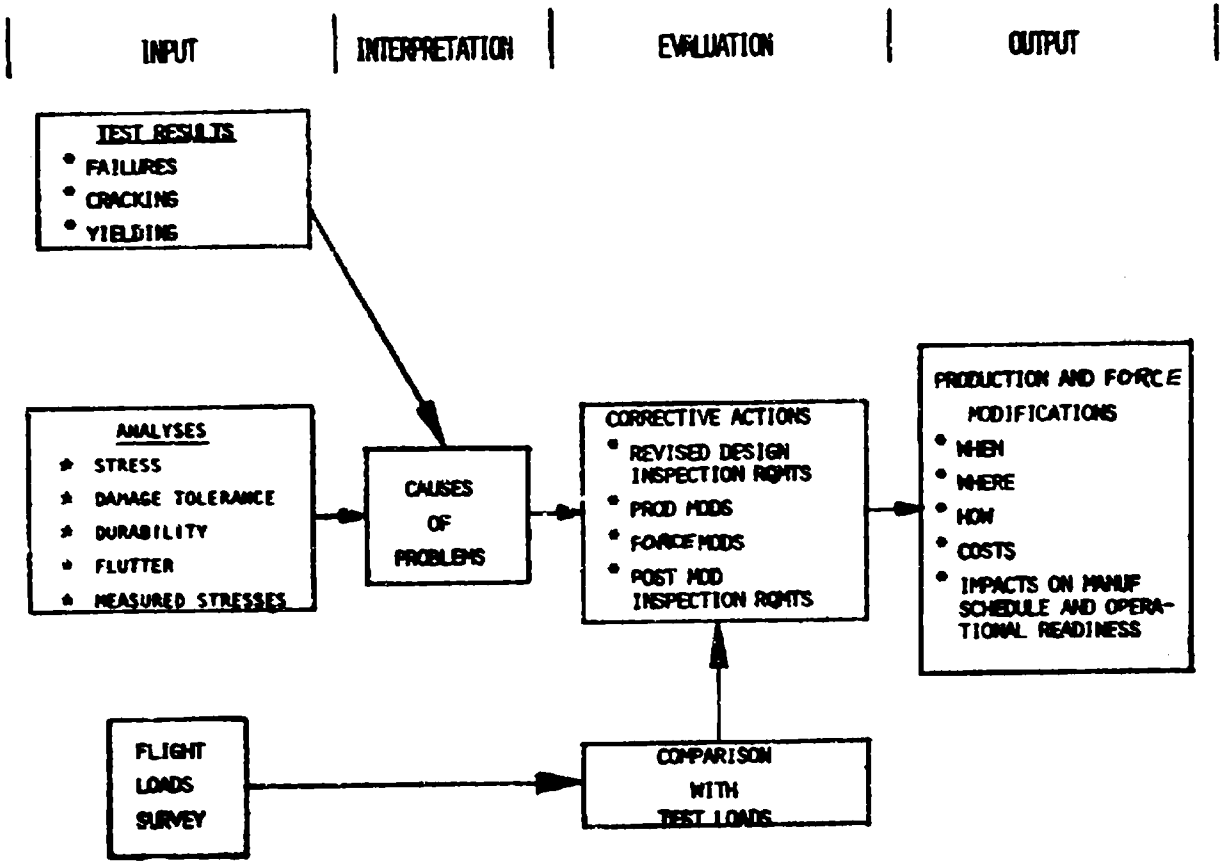

FIGURE 1. Interpretation and Evaluation of Test Results (Based on Design Service Life and Design Usage)

ALPHABETICAL INDEX

ALPHABETICAL INDEX (Cont'd)

ALPHABETICAL INDEX (Cont'd)

FIGURES

☆U.S.Government Printing Office: 1979 — 657-002/168

SPECIFICATION ANALYSIS SHEET	Form Approved Budget Bureau No. 22-R255

INSTRUCTIONS: This sheet is to be filled out by personnel, either Government or contractor, involved in the use of the specification in procurement of products for ultimate use by the Department of Defense. This sheet is provided for obtaining information on the use of this specification which will insure that suitable products can be procured with a minimum amount of delay and at the least cost. Comments and the return of this form will be appreciated. Fold on lines on reverse side, staple in corner, and send to preparing activity. Comments and suggestions submitted on this form do not constitute or imply authorization to waive any portion of the referenced document(s) or serve to amend contractual requirements.

SPECIFICATION

MIL-A-14350

ORGANIZATION

CITY AND STATE	CONTRACT NUMBER

MATERIAL PROCURED UNDER A

☐ DIRECT GOVERNMENT CONTRACT ☐ SUBCONTRACT

1. HAS ANY PART OF THE SPECIFICATION CREATED PROBLEMS OR REQUIRED INTERPRETATION IN PROCUREMENT USE?

A. GIVE PARAGRAPH NUMBER AND WORDING.

B. RECOMMENDATIONS FOR CORRECTING THE DEFICIENCIES

2. COMMENTS ON ANY SPECIFICATION REQUIREMENT CONSIDERED TOO RIGID

3. IS THE SPECIFICATION RESTRICTIVE?

☐ YES ☐ NO *(If "yes", in what way?)*

4. REMARKS *(Attach any pertinent data which may be of use in improving this specification. If there are additional papers, attach to form and place both in an envelope addressed to preparing activity)*

SUBMITTED BY *(Printed or typed name and activity - Optional)*	DATE

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